SEQUENCE EFFECTS IN PREDICTING FATIGUE DAMAGE OF HIGH PERFORMANCE AIRCRAFT

John E. Kane

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THESIS

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by

John E. Kane

June 1974

Thesis Advisor: G. H. Lindsey

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Sequence Effects in Predicting Fatigue
Damage of High Performance Aircraft

by

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TABLE OF CONTENTS

LIST OF TABLES

I. INTRODUCTION

The design phase of new aircraft involves the use of an estimated flight loading profile in order to make a service fatigue life estimate of the aircraft. A full-scale fatigue test is normally performed on the aircraft in the early stages of production to verify the usable life prediction; however it is inevitable that the actual loads the aircraft experiences throughout its service life will be different than the design spectrum. The fatigue monitoring program presently used for naval aircraft attempts to record, without regard to sequence, the actual loads an aircraft experiences by means of a four level exceedance accelerometer. Analysis of flight data from oscillograph recordings indicates that each type of aircraft has a distinguishable flight loading profile and it has been determined that the order in which these loads occur on a particular type of aircraft has a significant effect on the damage prediciton. This thesis proposes a procedure which estimates the actual flight profile from accelerometer data and statistically formulates a probable order in which these loads may have occurred. A fatigue cumulative damage theory is then incorporated which accounts for the sequential effects of loads in fatigue damage. The method is adapted to the computer which is necessary for monitoring the fatigue damage of a large fleet of aircraft and provides a damage calculation for each aircraft in approximately 2.5 seconds.

II. DAMAGE PREDICTION

Structural components which experience repeated loadings are susceptible to fatigue damage for loading amplitudes which occur well below the fracture strength of the material. These loading conditions occur frequently on aircraft, since the requirement of maintaining a particular flight pattern requires many loading changes. As a result, the design of aircraft structural components must account for the damaging effects of these repeated loadings, which necessitates an accurate means of predicting the fatigue damage accumulated due to these loading conditions.

Fatigue life will be understood to be the life until a visible crack is present, or the life until complete failure of a small component where the crack propagation phase would be short in comparison to the crack initiation period. Damage will refer to the percent of fatigue life expended due to a prescribed loading condition. Methods for predicting damage invariably rely upon some application of the time-tested Palmgrem-Miner Rule, better known simply as Miner's Rule.

A. MINER'S RULE

Both Palmgren and Miner are credited with introducing what has become the most well known incremental damage theory [Ref. 1]. If a component has experienced n_1 load cycles with the same mean load and load amplitude, the

portion of the fatigue life expended is n_i/N where N is the life to failure in a constant-amplitude test with the same mean and amplitude. Failure is assumed to occur if the sum of the consumed life portions equal 100 percent. This implies that the condition for failure is:

$$\Sigma n_i/N = 1 \tag{1}$$

This failure criterion was extended to a component experiencing load cycles of various amplitudes and means such that the condition for failure is:

$$\Sigma(n_i/N_i) = 1 \tag{2}$$

Specimen testing of different materials under varying mean and amplitude loads provide the N_i to use in Miner's Rule. The family of curves developed from specimen testing for a particular stress concentration factor are referred to as S-N curves.

1. <u>Advantages</u>

Due to the simplicity of application, Miner's rule has enjoyed widespread popularity as a means of predicting damage in aircraft. It is particularly suited to a large fleet of aircraft where loading data obtained from accelerometer readings can be readily stored in computers, which allows for fully automatic processing of data and evaluation of damage prediction for individual aircraft. Miner's rule

provides a quick "rule-of-thumb" indication of fatigue life expended since as the number of cycles of loading increases a greater portion of fatigue life is expended.

2. Limitations

Due to the complexity of aircraft loading cycles, Miner's rule has proven to be a rather conservative estimate of fatigue damage. For instance Lambert [Ref. 2] applied a typical aircraft block program to a rivetted lapjoint of 2024-T3 clad sheet aluminum. The ratio of actual life to Miner's estimated life obtained was 2.64. Through an extensive survey of the area of fatigue damage Schijve [Ref. 1] concludes that a Miner's Rule approximation provides a considerably inaccurate estimation of the fatigue life expended by aircraft. Economically it is not sound to retire an aircraft from service when actually only 37 percent of its fatigue life has been expended.

B. SEQUENCE EFFECTS

Since loading amplitudes vary randomly on an aircraft throughout its life, it has been determined that the order in which these loads are applied must be significant in the fatigue prediction. S-N data applied to Miner's Rule are based on data points obtained by specimen failure at a constant amplitude for a specific mean stress; therefore, it is highly unlikely that damage accumulation in aircraft components, which experience a wide range of amplitudes from cycle to cycle, would accurately adhere to a Miner's Rule

approximation. The order of loads is not taken into account by Miner's Rule. Schijve [Ref. 1] attributes the difference in damage accumulation between variable fatigue loading and constant amplitude loading to interaction effects.

1. Residual Stresses

If a load amplitude is of sufficient magnitude to cause plastic deformation in the region of a crack tip, residual stresses will remain in this region unless the deformation is fully reversed. These residual stresses have to be added to the stresses induced by the applied loads and as a consequence can significantly affect the fatigue damage accumulation.

2. Crack Closure Theory

If a cracked sheet is loaded in compression the crack will be closed. Elber [Ref. 3] observed that crack closure may occur while the sheet is still loaded in tension. According to Elber, plastic elongation will occur in the plastic zone of the growing crack. This plastic deformation will remain present in the wake of the crack and will cause closure before complete unloading of the specimen.

3. Application

Highly maneuverable aircraft experience a number of high g loadings followed by a large number of smaller positive g loadings. Prior to completion of crack initiation these high g loadings produce a residual stress at points of high stress concentration which effectively lower the mean values of the subsequent larger number of smaller positive g loads

and consequently reduce the actual fatigue damage. Once crack initiation is complete, these high g loadings produce residual stress in the crack tips, which tends to retard the propagation of the crack and consequently reduce the actual fatigue damage. If an extreme negative loading were experienced, the residual compressive stress induced would have the opposite effect and tend to cancel the favorable effects from the extreme positive loading. Figures (1a and 1b) demonstrate the results Schijve [Ref. 1] obtained from subjecting specimens of a known initial crack length to the different loading spectra. In Figure 1b note the significant number of additional cycles (point c) needed to obtain the same crack length after a few high positive loadings. The compressive residual stresses induced by high loadings seems to offer a plausible reason for the highly conservative estimates of aircraft fatigue damage from a Miner's Rule approximation. It should be noted that the damaging ground-air-ground cycle tends to relieve the positive effects of extremely high maneuver load residual stresses and should be account for in fatigue damage prediction.

In view of the effect that load sequence has upon fatigue prediction, it appears that not only are the loadings experienced by the aircraft important in damage prediction, but also the order in which they are imposed. A later section will discuss the Engineering Science Data Unit (ESDU)

Cumulative Damage Hypothesis, which takes into account sequence effects. The next section will address the problem of estimating the actual load spectrum experienced by a given aircraft.

III. DATA ACQUISITION

In transport aircraft the gust loading spectrum is considered to be the most significant loading spectrum that contributes to fatigue damage. These loadings appear to be Gaussian in nature and occur symmetrically about the one-g level. Consequently, statistical methods such as the power spectral density are being employed to determine the spectrum of loads that these aircraft have experienced. However in high performance military aircraft, maneuver loads are considered the damaging loads for fatigue damage, and the gust loads are either disregarded or considered as a negligible contributor to fatigue damage. Leybold's attempt [Ref. 14] to describe the maneuver load spectrum from known probability distribution functions met with some success but was restricted to a particular type of aircraft. Data [Ref. 11] have shown significant differences in maneuver loads within aircraft of the same type as well as between aircraft of different types [Ref. 8]. The main reason for this difference is the wide range of missions performed by these aircraft and yet knowledge of the loads experienced by individual aircraft is necessary for accurate damage predictions.

Stresses at fatigue critical points and their number of occurrences are required for fatigue damage prediction. Stress at a point cannot be recorded directly and consequently parameters that relate to stress must be utilized.

The most widely used parameter for calculating stress is strain. The theory of elasticity has mathematically developed the relationship of stress to strain. Unfortunately previous attempts to record strain history from strain gauges have been unsuccessful due to the unreliability of gauge readings caused by zero drift and the difficulty in processing and reducing strain data for a large fleet of aircraft. Scratch gauges have also been recommended and are capable of recording many hours of strain on a tiny disk but have the same logistics and reliability problems as strain gauges. Despite previous setbacks, strain recording devices are still considered to be the best means of obtaining stress. The National Aerospace Laboratory, Netherlands has developed a prototype system for in-flight strain recording with on-board data reduction [Ref. 4]. In this system the output of an ordinary resistance strain gauge bridge is amplified and digitized. This digital record is next analyzed for peak values using a "threshold" criterion. The values of successive minima and maxima are written on an on-board magnetic tape recorder. The prototype is currently installed in an operational F-104G aircraft for long-term operational testing. Initial results are highly promising.

Another method for determining stress, and the most widely used parameter for calculating stress on aircraft, is acceleration. Depending on the model of the accelerometer, discrete levels of g's exceeded are digitally recorded. Stress analysis of the fatigue critical points provides a relationship

between acceleration and stress. This method is presently used by the U.S. Navy.

A. COUNTING METHODS

Actual load-time histories will consist of a number of load excursions with an irregular pattern and in an irregular sequence. The analysis of load-time histories has to be such that the amount of damage caused by these load excursions is quantitatively reflected in the final results. Reference 5 describes ten different counting procedures and their ability to record significant fatigue loads on aircraft. The methods are subdivided into three distinct categories depending on the parameters of the loading recorded. Basically the counting procedures record either peaks (maximum or minimum values in Figure 2), level exceedances (for example levels one or two in Figure 2), or ranges (for example the distance between level one and minus one). The preferred method of counting is the range-pair-range method where all ranges of significance are counted. The drawback to this method is the extensive data reduction required.

The restricted level-crossing count method is also considered an effective means of recording fatigue damage. Accelerometers used in fatigue analysis of aircraft employ this method of counting. A crossing of a level with positive (or negative) slope is not made until the load has also crossed a second lower (or higher) preset level in the opposite direction. Interpretation is hampered by

intermediate load cycles demonstrated in Figure 2. Two different load patterns are depicted but will produce equal counting results. However the loss of intermediate values is not a serious setback according to van Dijk [Ref. 5], who concluded that the majority of the load fluctuations of interest were the larger excursions that originate from the steady-flight level.

B. ACCELEROMETER LIMITATIONS

The relationship between stress and acceleration at the center of gravity is not as explicit as the relationship between stress and the strain measured at the critical point. The stress-acceleration relationship varies with weight, speed, altitude, and flight configuration. However, Lambert [Ref. 6] does not consider the variation critical for high performance aircraft since it can be assumed that all maneuver loads which will cause significant fatigue damage occur in the operational exercise of the sortie. Gust and maneuver loads that occur during the transit phase are assumed to be negligible by comparison. The stress-acceleration relationship is considerably simplified by a proper choice of an operational weight and flight parameters.

IV. FATIGUE MONITORING PROGRAM

The trend to extended operational service lives of military aircraft emphasizes the need for accurate fatigue damage calculations. Since protection against fatigue does not end with production delivery, subsequent responsibility includes further recording of service loads and environments, monitoring and reevaluation in circumstances of mission changes, inspection and maintenance, and structural alterations throughout the service life. Due to the importance of fatigue strength in extending the service life of military aircraft, the U.S. Navy has developed an extensive fatigue monitoring program which presently monitors the individual structural fatigue status of 2700 aircraft. Expansion of the program is planned to monitor structural fatigue life expended on all Navy aircraft with the exception of transport and rotary wing aircraft.

The Naval Aircraft Structural Fatigue Life Program enables structural integrity decisions to be made realistically in terms of fatigue experienced by the structure rather than by continuing the previously nebulous procedure of using the number of years in service or the accumulation of flight hours as an expended life criterion. The Naval Air Development Center is tasked with monitoring the structural fatigue life expended on each Naval aircraft. As a result a fleet-wide counting accelerometer program, combined with pilot's

reports relating flight usage data, laboratory fatigue test data, and cumulative fatigue damage theory, was developed to provide a measure of the actual service life expended for an individual aircraft throughout its service life.

A. UTILIZATION

Usage of the fatigue data accumulated has provided the Navy with the source of information necessary to effectively manage a large fleet of aircraft. The data are reported quarterly [Ref. 6] and used for:

Early recognition of changes in service usage trends.

Identification and monitoring of aircraft accumulating unusually severe or excessive load factor occurrences and aircraft with unusually high fatigue damage accumulation rates.

Determination of aircraft time of retirement or rotation into or out of severe service usage.

Extension of service life based on changes in mission, flight restrictions, replacement or modification of major structural components, review of service history, and performance of full scale fatigue tests.

Provision of information effecting both organizational and in-service maintenance problems.

Utilization of data in the development/modification of naval aircraft design and test specifications/requirements.

B. DAMAGE PREDICTION

As was previously mentioned, the main function of the structural fatigue program is to monitor the damage accumulation of individual aircraft. Typical of the method used to arrive at the damage calculation is the method employed to calculate the damage on A-7B aircraft. The following discussion will specifically apply to A-7B aircraft but is considered representative of the other types of aircraft monitored under the program.

1. Full Scale Testing

The first step in developing a damage calculation on the A-7B was to determine fatigue critical points. These points were analyzed and S-N data appropriate to these points were developed. A block program based on estimated loading per 100 hours of life time was used to fatigue test the aircraft (Table I).

TABLE I

A-7 Block Loading

Maximum (g's) Load (g's)	Number of load applications per load block	
	Block 1-4	Block 5-10
2.45	1700	1700
3.15	950	950
3.85	650	650
4.55	450	450
5.25	250	250
5.95	136	136
6.65	44	44
7.35	15	15
8.05	4	4
8.75	1	2

Based on the proposed block loading program an estimated fatigue life was predicted at each critical point based on Miner's cumulative damage theory. The full-scale fatigue test was then conducted (using blocks 1-10, which represent 1000 hours of flight time) until a crack initiated at one of the critical points. The stress concentration factor of the S-N data was then adjusted to predict a fatigue life of 100 percent for the number of cycles that occurred on the full-scale fatigue test. The loads obtained from data reduction of accelerometer readings are then applied to the S-N data and a cumulative damage prediction is calculated. Due to the uncertainty of the actual flight loading history a safety factor of two is applied to the damage prediction.

2. Load Data Reduction

The accelerometers used on the aircraft record the total exceedances at only four preset levels. The custodian squadrons report quarterly to NADC the accelerometer readings and pertinent flight data. Based on the four exceedance levels recorded, seven other acceleration levels are computed for the loading spectrum to be used for fatigue prediction in the case of A-7 aircraft. The computed exceedances are determined by use of the following equation [Ref. 8]:

$$\text{Cumulative Counts} = Ae^{B/C} \tag{3}$$

Table II provides the values of the constants to be used for the seven levels.

TABLE II

A-7 Load Level Approximation

Cumulative Counts	A	B	C
5.5 g's	1.07	ln(5-g counts)+ln(6-g counts)	2
6.5 g's	1.00	ln(6-g counts)+ln(7-g counts)	2
7.5 g's	1.00	ln(7-g counts)+ln(8-g counts)	2
8.5 g's	1.00	2ln(8-g counts)-ln(7.5-g counts)	1
4.5 g's	.82	3ln(5-g counts)-ln(6-g counts)	2
4.0 g's	.97	2ln(4.5-g counts)-ln(5-g counts)	1
3.5 g's	.95	2ln(4-g counts)-ln(4.5-g counts)	1

The basis for these equations is obtained from reduced oscillograph data. An oscillograph recorder system is installed on a number of fleet aircraft which records time histories of airspeed, altitude, and normal acceleration. The recorder is energized by a switch on the landing gear retraction mechanism and consequently records data only when the landing gear is in the up position (the same is also true of accelerometer data so no landing information is provided). Periodically these oscillograph data are compiled on aircraft [Ref. 8] and used to develop relationships between the various levels. The data are also used to redefine the stress-acceleration relationship since altitude and airspeed data are correlated with the loads. Unfortunately a semi-automatic data reduction system is employed which outputs

data on all A-7's, and consequently, information on individual aircraft is not available for a comparative analysis of trends between individual aircraft. The 11 levels of acceleration now available are transformed into stress levels and the S-N data are referred to for damage prediction. The entire process has been adapted to a computer system which enables rapid retrieval of pertinent flight and fatigue data of any aircraft currently under this program.

C. IMPROVEMENTS

The fatigue monitoring program conducted by NADC has unquestionably made significant advancements towards an accurate prediction of fatigue damage. There are possibilities for improvement based upon current knowledge of the fatigue phenomenon.

It has been determined that maneuver loads occur randomly and that block testing will not produce the same fatigue damage as random load testing. Schijve [Ref. 1] mentions that, in general, the life in "equivalent" program tests is longer than in random tests. He alludes to one series of programs, where fatigue lives were about six times longer than in random load tests.

Through block loading fatigue testing on an A-7 aircraft, Miner's Rule has been altered by shifting the S-N data to account, somewhat, for the variation of load amplitude throughout the flight profile. Where Miner's Rule may be too conservative, block loading on the other hand may be too optimistic.

The sequence effects of random loading cannot be ignored if accurate predictions are to be obtained. The remainder of this thesis considers the feasibility of applying sequential effects for fatigue analysis based on four-level accelerometer readings and the suitability of applying this method to a computer. The next section will address the problem of developing a reasonable flight profile suitable for application to sequence effects and based on four levels of acceleration provided.

V. DEVELOPMENT OF A FLIGHT PROFILE

In order to apply sequential effects a reasonable flight profile must be produced that satisfies the following requirements:

1) The order in which the loads are encountered must be randomly distributed.
2) The loading spectrum must be generated from the four levels provided by the accelerometer readings.
3) Recognizing the individuality of aircraft flight profiles, two equivalent sets of accelerometer readings must not necessarily generate the same loading spectrum.

Reference [8] and other similar reports from NADC containing oscillograph information on aircraft were analyzed in developing a flight profile suitable for high performance military aircraft.

A. OSCILLOGRAPH/ACCELEROMETER DATA

Schijve [Ref. 1] in Figure 3 differentiated between the gust spectrum and the maneuver spectrum experienced by aircraft. The gust load spectrum consists of an equal number of positive and negative small amplitude peaks occuring about the steady flight condition. The maneuver load spectrum consists of the remaining loads an aircraft experiences and is highly asymmetrical about the steady flight condition. Lambert [Ref. 6] made a similar analogy as shown in Figure 4

and attributed the negative loads in the maneuver spectrum as being due to "overswing" resulting from recovery from a high-g maneuver. From an extensive study of Swiss VENOM aircraft Branger [Ref. 9] concluded that the number of maximum peaks above the steady flight condition was equal to the number of minimum peaks below the steady flight condition but that the distribution is highly asymmetric. Oscillograph data [Ref. 8] support this phenomenon. A lower threshold of two g's in the positive loading regime and an upper threshold level of .5 g's in the negative flight regime produced ratios of positive to negative g's as high as eight to one. The reason for this is demonstrated in Figure 3 where the number of positive loadings above threshold level A clearly exceed the number of negative loadings below threshold level B. Schijve [Ref. 1] indicates that all significant fatigue damaging loads originate from around the one-g level. Based on this evidence all accelerometer readings are assumed to be maximum peaks and likewise all oscillograph peaks in the positive loading regime are also assumed to be maximum peaks since the criterion for recording a peak is essentially the same as the accelerometer criterion. All loadings are assumed to originate from the one-g level or less (minimum peaks) depending on the number of negative regime peaks the oscillograph data provides.

B. ANALYSIS OF DATA

Different assumptions are made relating the peaks to a common trend and the data are analyzed to verify the validity of each assumption.

1. Cumulative Loads Between Types

Since all the aircraft considered were high performance aircraft, having missions that required high maneuvering loads, it was assumed that all aircraft experienced essentially the same number of loads based on 1000 hours of flight time. Figure 5 indicates the logarithm of the number of loads that exceeded a particular g level for A-4E, A-7A, F-4B, A-6A and F-8E aircraft per 1000 hours of flight time. The data would not support the assumption that the maneuver loads of each type of aircraft were essentially the same and thus the mission requirements of each type of aircraft are not comparable enough to produce an equal number of similar loads. However, due to the similarity of the curves the relative ratios of load levels between different types of aircraft may be proportional.

2. Ratio of Loads Between Types

Although all the aircraft considered had missions demanding high maneuver loads, some aircraft had missions that demanded more loads per flight than those of another aircraft type. In spite of this it was assumed the loading spectra were similar and the ratios of loading levels between the types of aircraft are essentially equal. All loading levels were ratioed to the lowest positive load

level recorded. Figure 6 shows the ratio of five load levels for A-4E, A-6A, A-7B, F-4B, and F-8E to their respective 2.0-2.5 g load counts. Two of the aircraft's ratios deviated considerably from the other three aircraft's ratios. Intermediate load levels were then investigated to avoid the possible effect of extreme gust loads at the lower level. Figure 7 shows the ratio of four load levels to the 4-4.5 g load level. The ratios again indicated a considerable degree of scatter. All loads that exceeded an upper level were ratioed to the load counts at lower levels. Figure 8 shows the ratio of four load levels to the number of loads that exceeded the four g load level. These ratios indicated even more scatter than those obtained in Figure 7.

Since the load ratios investigated did not indicate any clear similarity, it was concluded that the flight profiles between types of aircraft were distinctly different.

3. Extreme Values

Extreme values were investigated to determine whether there was a relationship between the load exceeded by extreme positive and negative loads. It was assumed that entry into these extreme loads was not intentional and probably flights that encountered these loads encountered extreme values in both directions. Extreme values were defined as loads that accounted for one percent of the total number of loads occurring in either the positive or negative regime. Table III lists the number of positive and negative levels exceeded one percent of the time.

TABLE III

Extreme Value Occurences and Level of Exceedance

A/C Type	Level	No. Exceed.	Level	No. Exceed.
A-4E	6.0	282.6	-.41	17.2
A-6A	5.5	88.9	-.21	4.9
A-7A	5.5	177.5	-.1	20.3
F-4B	5.5	42.1	-.41	8.1
F-8E	6.0	117.8	-.41	35.4

Analysis of Table III indicates no definite relationship between the level exceeded in the positive regime and the level exceeded in the negative regime. A sample correlation coefficient of .642 for levels of extreme-value exceedance was not considered conclusive enough to support the assumption that a relationship existed between the positive and negative levels. Analysis of extreme values on a flight-by-flight basis is needed to confirm the assumption.

Reference 10 contains oscillograph data on the Navy's "Blue Angels" Flight Demonstration Team compiled on a flight-by-flight basis. These data were also analyzed for a possible trend. By analyzing the data on an individual flight basis it could be determined whether or not extreme high loads tend to induce corresponding extreme low loads as a result of over-correcting recovery from the high load maneuvers. Table IV lists the highest and lowest g level recorded for 12 flights analyzed at random.

TABLE IV

Blue Angel's Flight-by-Flight Extreme Values

Flight	Max. g	Min. g
1	7.4	- 1.56
2	6.18	- 1.05
3	5.83	- 1.41
4	4.6	- 1.56
5	5.62	- 2.87
6	6.48	- .11
7	6.48	- .05
8	6.15	.04
9	7.38	- 1.27
10	6.63	.08
11	5.22	- 1.15
12	6.49	- 1.91

Detailed analysis of Reference 10 indicated that extreme low loads neither preceded nor followed an extremely high load and as Table IV indicates there is absolutely no foundation for assuming a high probability of both high and low values occurred on the same flight. Of particular significance in Table IV is the fact that the lowest negative reading (-2.87) occurred during a flight which had the third lowest positive reading. It must be concluded that the negative readings occur randomly and independently of any extreme high loading conditions; consequently, they could be considered as part of a "basic" loading spectrum unique to the flying qualities of the particular aircraft considered. The extreme low g

loads are probably due to high gusts loads and therefore the gust loads should be considered integrated into the "basic" loading spectrum. The dynamic "overswing" Lambert [Ref. 6] discussed must be above the .5 g level threshold. From the data investigated so far it appears that each aircraft has a flight loading profile distinct from another type of aircraft and this profile includes all the negative g loadings and excludes all high g loads. If types of aircraft do indeed have a loading spectrum "fingerprint", it can only be determined by comparing oscillograph data of individual aircraft of the same type.

4. Individual Aircraft by Type

In order to substantiate whether aircraft of the same type have identical flight profiles below some extreme positive load level, individual aircraft data are needed. A study conducted for NADC by Technology Incorporated [Ref. 11] contains individual aircraft oscillograph data on four A-6A's and six A-7B's collected over a one-year period and representing data on 306.9 flight hours for the A-6A's and 501.9 flight hours for the A-7B's. Although the number of aircraft of one type analyzed was small it was considered significant enough to determine if aircraft of the same type do experience essentially the same loading profile below some high load level.

a. Validity of Data

The data were first compared with the oscillograph data of Ref. 8 to assure comparable results between the two

reports. The A-6A data agreed quite closely and the A-7B data also agreed favorably in the lower loading regime but diverged significantly above the five-g level. This was not considered important in this investigation, since the region of interest for developing a loading profile is contained below the five-g level. Since the ratios of loads were considered to be the key to developing a flight profile, the data were considered to agree favorably with other oscillograph data of the same type aircraft.

b. Comparisons of Each Type

The number of exceedances for individual aircraft of both types at each level were compared. The exceedance distribution for individual aircraft of each type indicated a high degree of similarity which would be necessary to develop a flight profile for each type. Figure 9 shows the number of loads exceeding an indicated level per 1000 hours of flight time for each of the A-7B aircraft. The negative loads also exhibited a comparable similarity. Analogous results were also obtained for the four A-6A aircraft.

The number of loads occurring in each range were ratioed to an arbitrary level that had a range of one g. The one-g range level was chosen to agree with the ranges between recording levels of accelerometers installed in the aircraft. The number of loads that occurred within a prescribed range were determined by subtracting the number of exceedances at the higher level of the range from the number of exceedances at the lower level of the range. The-one g range level was

increased and comparisons made until an unacceptable disagreement resulted between individual aircraft. Acceptable agreement for the A-6A occurred only for readings up to the 3-4 g range. However acceptable agreement for the A-7B occurred up to the 4-5 g range. Figure 10 shows the ratio of load occurrences to the number of 4-5 g occurrences for the A-7B aircraft. The load values were represented by the mean value of each range. The fact that one aircraft experienced an increase in the ratio from the 2.5-2.9 range to the 2.9-3.3 range was not considered significant since that aircraft had recorded only 19.7 hours of flight time with the oscillograph installed. It was considered significant that four of the six aircraft had almost identical values.

c. Flight Profile

Based on the above results it was decided to develop a flight profile for A-7B aircraft that would depend solely on the accelerometer counts in the 4-5 range. Table V indicates the load range and the ratio of that range to the 4-5 g's each aircraft experienced. The ratio of loads in each range were assumed to be normally distributed about the mean ratio. Based on these data a mean value and a sample standard deviation were calculated. It was decided that a range of 1.65 times the sample standard deviation from each ratio mean would encompass an adequate range of loads that the aircraft would experience in each load level. Multiplying

TABLE V

A-7B Individual Ratios of Positive Loads to 4-5 g Loads

Load Range	19.7 hrs.	50.2 hrs.	140.5 hrs.	101.3 hrs.	113.1 hrs.	76.9 hrs.
2.0-2.5	3.74	3.81	3.29	5.24	5.14	2.85
2.5-2.9	1.87	1.67	1.86	2.67	1.76	1.73
2.9-3.3	1.99	1.18	1.46	2.51	1.27	1.50
3.3-3.6	1.14	.658	.823	1.49	.714	.855
3.6-3.9	.73	.504	.621	.770	.559	.605
3.9-4.2	.444	.372	.438	.414	.400	.439
4.2-4.5	.298	.333	.339	.304	.347	.325
4.5-4.8	.275	.268	.252	.275	.257	.251
4.8-5.0	.152	.149	.143	.181	.143	.135

each ratio by the number of 4-5 g readings would produce the actual number of loads the aircraft experienced for that flight period. 3.3 times the sample standard deviation provides a probability of .95 that the aircraft experienced a load level in that range based on a sample size of six. Table VI gives the range of ratios for each load level that the aircraft is allowed to experience in the positive regime for each 4-5 g reading.

Table VII gives a similar list of ratios as Table V for the negative regime. The excessive number of negative readings aircraft (Serial No. 154370) experienced were not considered in the calculation of mean and sample standard deviation of the negative regime ratios. This aircraft had only 19.7 hours of recorded load time and its ratios were

TABLE VI

A-7B Positive Ratio Range For Each Load Level

Range Mean	Low	Mean	High
2.25	2.399	4.01	5.62
2.7	1.306	1.92	2.53
3.1	.815	1.65	2.48
3.45	.429	.947	1.47
3.75	.465	.6315	.798
4.05	.370	.417	.464
4.35	.292	.324	.356
4.65	.245	.263	.281
4.9	.122	.1505	.179

TABLE VII

A-7B Individual Ratios of Negative Load Range to 4-5 g Load

Load Range	19.7 hrs.	50.2 hrs.	140.5 hrs.	101.3 hrs.	113.1 hrs.	76.9 hrs.
.6-.4	18.97	9.02	4.807	9.24	6.74	5.97
.4-.2	4.66	2.96	1.506	1.88	1.412	.606
.2-.0	1.22	.601	.376	.408	.329	.080
0-(-.2)	.274	.175	.137	.110	.071	.020

not considered representative of the typical A-7B. Neglecting the excessive number of negative loads on this aircraft would not be more damaging in a cumulative damage calculation since sample calculations with representative S-N data indicate a

loading with a larger range but a lower mean was less damaging than a loading with the same maximum peak that originated at one g and had a smaller range. Table VIII indicates the ranges of ratios in the negative regime that have a probability of .95 of occurring for each 4-5 g reading based on a sample size of 6.

TABLE VIII

A-7B Negative Ratio Range For Each Load Level

Range Value	Low	Mean	High
.538	3.97	7.155	10.34
.346	.260	1.673	3.09
.148	.050	.359	.667
- .054	.004	.103	.200

Figures 11 and 12 display the range limits and mean for each load level for both the positive and negative regime, respectively. The mean value of each positive load level is considered to be an adequate representation of the loads in that range since the greater number of loads that occur below the mean value are offset in the damage calculation by the fewer but more damaging high values in each range.

5. Sequence of Loads

Based on the previous constraints a computer program was developed to arbitrarily assign a ratio for each load level to the 4-5 g level reading. The value of each ratio

was determined by means of a normally distributed random number generator. The number of actual loads in each load level were then determined by multiplying each ratio by the number of 4-5 g readings that were inputted.

Three other levels of loads from accelerometer readings were also inputted that represented loads that occurred in the 5-6, 6-7, and 7-8 load ranges respectively. These load ranges were arbitrarily assigned mean values of 5.75, 6.6, and 7.8 which were considered high enough to represent a conservative estimate for a damage calculation of a large number of loads occurring in this range since 75 percent of loads recorded occurred below these values in each range [Ref. 11]. No attempt was made to distribute these high level loads throughout the entire range of their possible values due to the limited number of occurrences of these loads in Ref. 11. However if distribution functions for the extreme value loads were known it would be appropriate to subdivide the loads occurring in each range into a smaller number of load ranges. Distributions similar to the one used in Table II could be used but in this particular case the distribution could not be substantiated from the data in Ref. 11.

Based on the four load level accelerometer readings, all loads in the positive regime were assigned to be maximum peaks. Associated with the maximum peaks was an equal number of minimum peaks which were assigned values from the negative

regime and the remaining number of required minima were assigned a value of one g.

A uniformly distributed random number generator placed each group of peaks in a random order. Starting with a minimum peak, a load value was picked from each group until all the loads in each group had been chosen. A final half cycle was then assigned to finish at a one-g level. Figures 13-16 indicate four flight profiles generated for accelerometer readings of three, five, three, and four representing loads occurring in the 4-5, 5-6, 6-7, and 7-8 g ranges respectively.

It is highly unlikely that the order of loads generated from the accelerometer readings would represent the actual sequence the aircraft experienced during that period and thus a damage calculation based on this loading profile would be inaccurate. However, as the aircraft experiences an increased number of loadings over its fatigue life it is highly probable that the aircraft experiences a majority of the sequential loadings generated or ones quite similar to them since the loads generated are based on actual data obtained from aircraft of the same type whose order of loading is known to occur randomly. The flight profiles generated will be of increasing significance during the latter stages of the aircraft's life where the accuracy of a damage prediction calculation becomes of increasingly higher importance.

6. Suitability of Flight Profile

The requirements stated previously for generation of a profile have been met. The loads are randomly distributed and are generated solely from accelerometer readings. Two equivalent accelerometer readings do not necessarily generate the same loading profile. Due to the small number of aircraft sampled it cannot be assumed that the above profile is an accurate representation of the majority of A-7B's profiles. A larger sampling of aircraft would be necessary to further refine a representative profile. Currently the lowest accelerometer level being recorded on A-7B's is the five-g level. In order to adapt the flight profile developed, the lowest level recorded would have to be lowered to the four-g level. The same is also true of the A-6A where the lowest level presently recorded is the four-g level and a three-g level is considered necessary to obtain an indicator of the A-6A profile. However the above procedure demonstrates the feasibility of generating a flight profile from accelerometer readings since each type of aircraft apparently does have a "fingerprint" load spectrum below some high load level. The flight profile developed above will be considered representative enough of high performance fleet aircraft to proceed with accounting for the sequential effects in a fatigue damage calculation. The next section deals with the feasibility of applying a fatigue damage prediction that accounts for sequential effects and is suitable for computer analysis.

VI. ESDU CUMULATIVE DAMAGE HYPOTHESIS

A cumulative damage hypothesis is needed that accounts for the order in which loads are applied to an aircraft that is suitable for computer analysis.

The ESDU Cumulative Damage Hypothesis [Ref. 2] meets these requirements. Although originally intended for block loadings the method has been modified for analysis of loading on a cycle by cycle basis. The ESDU method requires no additional data over that required for use in Miner's Rule. Testing [Ref. 2] of lab specimens under block program loading has produced results that demonstrated a far better prediction of fatigue failure by the ESDU method than by Miner's Rule. A number of similar specimens were tested until failure which on the average took 30.9 programs to occur. The Miner's Rule estimate for failure was 11.7 programs; whereas the ESDU estimate was 28.7 programs, an improvement of over 240 percent.

The basic hypothesis of the ESDU method is centered in the concept that once the stress applied to a component exceeds the proportional limit, residual stresses build up which effectively alter the actual mean stress of subsequent stress cycles. The following basic assumptions are made when applying the ESDU method:

1) The material is perfectly elastic below the yield stress and perfectly plastic above it. The yield stress is taken as the .2 percent offset stress.

2) The stress-strain characteristics of the material are the same in tension and compression.

3) The stress concentration is constant for all nominal stresses.

4) Yielding is localized to stress concentrations and therefore does not significantly affect stresses in the rest of the member.

5) A constant amplitude S-N curve for the material is available.

A member that is subjected to a nominal stress whose peak value is SMAX has an elastic stress at the stress concentration of SMAXEL equal to:

$$\text{SMAXEL} = \text{KT} \cdot (\text{SMAX}) \tag{3}$$

If the value of SMAXEL exceeds FP, the .2 percent offset yield stress, the mean value of the subsequent cycles will be lowered by an amount equal to H where:

$$\text{H} = \text{FP} - \text{SMAXEL} \tag{4}$$

The same is also true for an excessive compressive stress, which due to the build up of residual tensile stress causes an effective raising of the mean stresses of the subsequent cycles. Figures 17-19 indicate the resulting effect that the different types of loadings have on subsequent cycles. In Figure 17 neither the maximum nor the minimum peak exceed

FP so no alteration of the mean stress in subsequent cycles is required. Figure 18 indicates the effective raising of the mean stress of subsequent cycles due to a compressive stress that exceeded -FP. Figure 19 shows the effect of compressive residual stresses due to a tensile loading whose maximum value exceeded +FP.

Application of the ESDU method is quite simple and is easily adapted to a cycle-by-cycle analysis. A possible objection to the validity may be that the effects of stress relaxation are neglected, which causes the residual stress to fade away if given enough time. However Schijve [Ref. 1] states that this effect has not been observed in aluminum alloys, the materials most aircraft wing components are made from.

The next section discusses the computer program used to develop a flight profile from accelerometer readings and to apply the ESDU Cumulative Damage Hypothesis for a damage calculation of this profile.

VII. DISCUSSION OF COMPUTER PROGRAM

Program ESDU was developed primarily to demonstrate the feasibility of adapting the ESDU Cumulative Damage Hypothesis to a computer program for use in calculating the fatigue damage of high performance aircraft with accelerometer readings as data input. In order to provide a concrete example the program was specifically designed to provide damage calculations on A-7B aircraft that had accelerometer recording levels at the four, five, six, and seven g level.

A. FEATURES

There is virtually no limitation to the number of readings at each level (ten digit maximum); however, only a limited number of readings will be used in the calculation if the number of four-g level readings exceeds 325. The program requires 152 K bytes of storage and less than three seconds of computer time for each aircraft calculation. Data input consists of the number of aircraft to be analyzed, the serial number and four accelerometer readings for each aircraft.

B. PROCEDURES FOR CALCULATION

The program is essentially subroutine structured with the main program linking the ten subroutines required for the calculation. The main program accepts the input data, assures compatible storage requirements, requests maximum

and minimum peaks, requests a random ordering of the peaks, and finally calls for a calculation of damage for the flight sequence using the ESDU method. A brief description of each subroutine follows.

1. Subroutine Fix

Fix assures the storage requirements will not exceed 152 K bytes. If more than 325 four-g level readings are inputted for one aircraft, it is hypothetically possible to generate more than the 10,000 peak storage limitation. If more than 325 four-g level loads are inputted the four recorded levels are ratioed so that 325 four-g level loads will be provided to the main program. Fix also provides a correction factor to compensate for the reduced number of peaks in the final damage calculation.

2. Subroutine Upspec

Upspec provides the main program with the number of maximum peaks and their values to be used in the damage calculation. A ratio of each load level below four g's to the number of 4-5 g loads is determined by a normally distributed random number generator provided by subroutine Gauss. If the random number provided exceeds the .95 percentile range for each load level a new number is requested. After the number of loads in each level are determined, subroutine Hiload assigns the mean g load value to each peak in each load range.

3. Subroutine Dnspec

Dnspec performs essentially the same function as Upspec, except it provides the main program with minimum peaks. Appropriate ratios are assigned to four levels of minimum peaks. After determination of the number of peaks required in each load level the remaining required minimum peaks are assigned to a fifth level, which represents the one g minima, to equalize the number of maxima and minima. Subroutine Dnload assigns the load value to each of the five ranges.

4. Subroutine Mixer

Mixer provides the main program with the maxima and minima arranged in a random order so as to represent a flight profile. Mixer requests subroutine Shufle to randomly arrange both the maxima and minima by means of a uniformly distributed random number generator provided by subroutine Random. Mixer then alternately picks a value from both groups and stores them in an array until all peaks have been used.

5. Subroutine ESDU

ESDU calculates the damage incurred by the aircraft on a cycle-by-cycle basis using the ESDU Cumulative Damage Hypothesis. ESDU requires data about the particular aircraft being analyzed. A stress concentration factor, S-N data, ultimate tensile stress, .2 percent offset yield stress, and conversion from g to stress are needed for the aircraft at the fatigue critical point. Polynomial approximations

were used for the S-N data but more precise descriptions of the S-N data would be desirable. Linear interpolation of tabulated S-N data would be an acceptable method. S-N data for only one mean value are all that is necessary since the program adjusts the S-N data to accommodate different means by the method described in Ref. 2. In the next section sample data runs are discussed and the relevancy of their results to actual aircraft predictions.

VIII. DATA ANALYSIS

Actual accelerometer readings reported from fleet aircraft during heavy maneuver periods were fed into the program to determine if reasonable damage predictions would result. Although it was realized that the damage predictions have no realistic value except for A-7B's due to the flight profile variations between types of aircraft, subroutine ESDU was altered to accommodate the pertinent load data for A-6A and F-4B aircraft as well as A-7B aircraft. A fourth "hypothetical" aircraft was devised in order to cover a range of stress concentration factors and g-to-stress conversions commonly found in high performance aircraft. The stress concentration factors and g-to-stress conversions represent the actual values for the fatigue critical points of the aircraft types tested. The g-to-stress conversion is determined by the design limit load of the aircraft, which is the amount of nominal stress the fatigue critical component experiences at the 6.5 or 7.0 g load depending upon the aircraft in question. The design limit loads varied from 40,000 p.s.i. on the A-6A to 24,600 p.s.i. on the A-7B. S-N data was obtained by interpolation of Goodman Diagrams. contained in Ref. 12. The material used at the fatigue critical points for all aircraft was aluminum plate 7075-T6. Table IX provides the pertinent information on each aircraft used in the test runs.

TABLE IX

Aircraft Tested

Aircraft	F-4B	A-7B	A-6A	Hypo.
KT	3.3	3.5	3.0	2.5
stress (KSI) per g	4.723	3.5143	6.154	4.57
Location critical pt.	outer wing panel	outer wing panel	inner wing panel	-
Flight hours	25.8	31.5	33.4	31.5

Five test runs were performed on each aircraft to determine if the prediction variation was significant with different flight profiles. Data for the 4-5 g level of the A-7B were developed from the equations provided in Table III. Each identical flight profile was subjected to a Miner's Rule approximation for comparison with the ESDU method. Tables X-XIII show the results of these runs for each aircraft.

TABLE X

F-4B Test Runs with 25.8 Hours of Combat Flight Time

Acceleration data (g's)				Damage prediction	
Four	Five	Six	Seven	ESDU	Miner's
30	11	2	0	.0005824	.0156074
30	11	2	0	.0006568	.0153307
30	11	2	0	.0008384	.0157427
30	11	2	0	.0009822	.0157232
30	11	2	0	.0010484	.0155861

TABLE XI

A-6A Test Runs with 33.4 Hours of Combat Flight Time

Accelerometer data				Damage prediction	
Four	Five	Six	Seven	ESDU	Miner's
26	5	0	0	.0010131	.0147439
26	5	0	0	.0012068	.0147753
26	5	0	0	.0011503	.0165619
26	5	0	0	.0012042	.0158776
26	5	0	0	.0014054	.0166387

TABLE XII

A-7B Test Runs with 31.5 Hours of Combat Flight Time

Accelerometer data				Damage prediction	
Four	Five	Six	Seven	ESDU	Miner's
194	120	57	12	.0018248	.0118987
194	120	57	12	.0017034	.0116276
194	120	57	12	.0017585	.0117723
194	120	57	12	.0017833	.0119610
194	120	57	12	.0016321	.0116316

TABLE XIII

Hypothetical Aircraft with A-7B Flight Data

Acceleration data in g's				Damage prediction	
Four	Five	Six	Seven	ESDU	Miner's
194	120	57	12	.0016852	.0058465
194	120	57	12	.0014576	.0059167
194	120	57	12	.0014858	.0061357
194	120	57	12	.0013641	.0059771
194	120	57	12	.0012437	.0058356

In order to easily interpret the degree of variation between the five flight profiles, the mean value of the damage calculation and the percent of deviation from the mean of the largest excursion for each aircraft was calculated. The results are tabulated in Table XIV.

TABLE XIV

Results of Test Runs

Aircraft	F-4B	A-7B	A-6A	Hypo.
ESDU mean	.0008216	.0017404	.001196	.0014473
Percent excursion	29.11	4.847	17.5	16.44
Miner's mean	.015598	.011778	.015719	.0059423
Percent excursion	1.71	1.55	6.2	3.25

Some definite conclusions were made from the test runs. The ESDU method makes a considerable difference in a damage prediction as compared to Miner's rule. The variation from the mean in the damage prediction was noticeably larger in the ESDU method than from Miner's calculation, although they both were based on the same flight profile. Apparently the variation in the number of loads in the profile is not quite as significant in the damage calculation as is the order of the loading since the ESDU method, which accounts for the order of the loads, had a much larger deviation.

A. LIFE ESTIMATE

It was decided to calculate the life expectancy of an A-7B based on the number of occurrences of loads per 1000 hours from the oscillograph data of Ref. 8. Due to the large number of loads involved the data were subdivided into blocks of 26 representing 38.46 flight hours each. The loads were also divided into blocks of 52 representing 19.23 flight hours each. Ten runs were made on each set of data and by using the mean damage value each damage calculation was multiplied by the respective number of blocks to produce the damage accrued for 1000 hours. The same procedure was followed for a Miner's Rule prediction. The results are shown in Table XV.

Increasing the number of blocks from 26 to 52 resulted in an increase in the damage life for 1000 hours of 17.9 percent whereas the Miner's damage prediction showed a negligible increase of .396 percent. The reason for this

TABLE XV

A-7B Damage Accumulated for 1000 Flight Hours Based on Accelerometer Readings of 146, 64, 18, and 1 for 52 Blocks

Method	No. of Blocks	Damage/Block	Total Damage
ESDU	26	.0008126	.021128
ESDU	52	.000479	.024908
Miner's	26	.00405351	.1053913
Miner's	52	.0020348	.1058096

increase is due to the fact that everytime a block begins, the material is essentially free of residual stresses, and thus no favorable mean stress correction occurs until a high g load causes plastic deformation. The Miner's approximation is not concerned with the build-up of residual stresses, and thus should show no significant change in damage prediction as the number of blocks increases.

It was assumed that after every flight the favorable residual stresses were relieved by the damaging effect of landings and the taxiing phase. It should be noted that the taxiing phase will only have a damaging effect on fatigue critical points outboard of the main mounts. In addition it was also assumed that the damage prediction increased linearly as the number of blocks increased. Based on these assumptions, a damage calculation for the 1000 flight hours was divided into 500 flights that would represent an average of 2 hours between actual residual stress relieving. The damage calculated based on a linear relationship resulted in a

damage prediction of .09 or an expected life of 11,000 hours. This was not considered totally satisfactory since NADC's predictions are producing life estimates [Ref. 4] in excess of 22,000 hours (excluding the factor of safety).

B. EFFECTS OF STRESS RELIEF

It was decided to investigate the relationship between increasing flight blocks and damage prediction. The A-7B previously used for a one-block flight was divided into flight blocks of one, two, three, four, six, and twelve. The damage prediction was then calculated for the entire 31.5 hours of flight time. The results, including Miner's prediction, are indicated in Table XVI. Ten runs with each set of data were run with the mean value of the damage for the ten runs tabulated.

TABLE XVI

A-7B Damage Predictions with Increasing Flight Blocks

No. of Blocks	Damage/Flight	Total Damage	
	ESDU	Miner's	ESDU
1	.00174044	.01177824	.0017404
2	.00106697	.01215122	.0021339
3	.00070075	.0019376	.00210225
4	.00058844	.01185008	.00235376
6	.00043121	.0123125	.00025873
12	.000232184	.01176695	.0027862

Except for the decrease in prediction from a block of three flights the damage is clearly increasing as should be expected. The predictions from Miner's Rule are relatively stable. Although the increase in damage prediction is not linearly related to an increase in flight blocks, the relationship indicated a linear increase with the logarithm of the number of flight blocks. Figure 17 indicates the trend. Based on these results an aircraft damage prediction could be determined by using a one flight block run and an additional block run that would be compatible with input data. Then, assuming a linear relationship between the damage and the logarithm of flight block, the damage for any number of flights during that block could be determined. Based on this relationship the damage for the 500 flight block of the oscillograph data for 1000 flight hours produced a damage prediction of .0372 per 1000 hours or a life expectancy of 26,851 flight hours. This may seem a rather high estimate, but the loading profile the aircraft was designed for is considerably higher than the actual loads experienced.

It is not suggested that the linear relationship of the logarithm of the number of flights is clearly a linear relationship over the entire range. Extending predictions based on 26 or 52 flights for 500 flight blocks would not produce accurate results. However if the number of flights was close to the blocks used in obtaining two point predictions the linear relationship should produce reasonable results. In the case of the A-7B with 31.5 hours of

accelerometer data, blocks of one and twelve flights should predict a reasonable damage prediction for the aircraft if in fact 15 flights were involved during this reporting period which results in an average flight time per flight of 2.1 hours.

How much relief of residual stresses is caused by the landing and taxiing portion of the flight is presently unknown. Assuming that all the residual stress is relieved as a result of the ground phase is as unreasonable as assuming that none of the residual stress built up by a high load is relieved. Somewhere in between these two extremes is the actual amount of stress relief that occurs. Until this information is known the assumption that total stress relief occurs after each flight is a highly conservative estimate that would produce damage predictions that have a built-in factor of safety.

IX. SUMMARY AND RECOMMENDATIONS

A need has been demonstrated for the implementation of a damage hypothesis in fatigue prediction that accounts for sequential effects. Predictions based on block loading programs have shown to be an optimistic prediction of the fatigue damage. Recording devices that have a direct relationship to the stress experienced at a fatigue critical point such as time history strain gauges are highly desirable. Due to the nature of aircraft loadings, most fatigue damaging loads are initiated from the one-g level or slightly below it. As a result of this type of loading, accelerometers were considered acceptable for indicating the number of loads that occurred above the preset levels. Unfortunately no information was provided concerning the order in which the loads occurred. It was found that the occurrence of a particular load in flight could not be sequentially associated with the occurrence of any other load. As a result it was concluded that maneuver loads occur randomly.

Based on accelerometer data, a flight profile was developed that includes all fatigue damaging loads that a particular type aircraft experiences. Unfortunately the load levels now being recorded on the A-6A and A-7B aircraft are above the level necessary to obtain the flight profile.

The need for recording load levels above the eight-g level in A-7B aircraft is appreciated due to the highly

damaging effect these loads impose; however, analysis of flight data indicates that loads above the eight-g level occur on the average less than five times for every 1000 hours of flight time. Reference 13 indicates that loads with a peak of 8.05 g's produce a damage of .0001. Based on these data an A-7B which experienced only 8.05 g's could fly for 2,000,000 hours before fatigue failure. It is recommended that the accelerometer levels on A-7B aircraft be lowered to include readings at the four-g level. Estimation of eight-g occurrences could be provided by allowing a reasonable number of the seven-g loads to occur above the eight-g level by means of some distribution similar to the equations provided in Table III.

Randomizing the order of flight loads has a significant effect on the damage calculation provided by the ESDU method. During the life span of the aircraft, it is anticipated that the majority of the loading sequences generated would have actually been experienced by the aircraft. Varying the ratio of loads in each range had a less significant effect on the damage prediction, since the Miner's Rule approximation produced approximately similar results regardless of the number of loads a particular trial run produced. Neglecting a variation of load level ratios would preclude the possibility of one aircraft accidentally being assigned an extreme low ratio for each flight period.

The flight profile generated from a small sample of aircraft cannot be considered as conclusive evidence that

each aircraft has a "fingerprint" flight profile below some high load level. Oscillograph data now compiled by type of aircraft should be reduced on an individual aircraft basis so that the profile hypothesis can be substantiated.

The ESDU cumulative damage hypothesis provides a reasonable explanation of the overconservativism of the Miner's approximation. The ESDU method is suitable for computer analysis of damage prediction of a large fleet of aircraft due to the short run time required for calculation.

Assuming that the residual stress imposed on the aircraft component due to a high load would remain indefinitely is unreasonable. However the amount of stress relief caused by landing and taxiing loads is not available. It is recommended that the amount of stress relief imposed by the ground phase of the flight be given considerable attention to enable an increased accuracy of damage prediction by the ESDU method. Until information is known concerning the ground phase of the flight, each flight could be assumed to relieve completely the favorable residual stresses that developed during the flight.

APPENDIX A

PROGRAM ESDU

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CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C                   PROGRAM ESDU UTILIZES THE ESDU
C                   CUMULATIVE DAMAGE HYPOTHESIS FOR
C                   PREDICTING THE EXPENDED FATIGUE
C                   LIFE OF A-7B A/C FROM ACCELEROMETER
C                   READINGS AT THE 4,5,6, AND 7 G LEVEL
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      DIMENSION UPPEAK(5000), DNPEAK(5000), SEQUES(10001)
      WRITE (6,6)
      READ (5,7) NUM
C
      DO 5
      READ (5,8) NACRFT,FOUR,FIVE,SIX,SEVEN
      AJUST=1.0
      PREDIC=0.0
      IF (FOUR.GT.325.0) GO TO 1
      GO TO 2
    1 CALL FIX (FOUR,FIVE,SIX,SEVEN,AJUST)
C
C                   ASSURES MEMORY REQUIREMENTS SATISFIED
C
C
C                   SUBROUTINE TO GENERATE MAXIMUM G PEAKS
C                   A/C HAS EXPERIENCED
C
    2 CALL UPSPEC (NPEAK,NACRFT,FOUR,FIVE,SIX,
     1SEVEN,UPPEAK,IIII)
      IF (NPEAK.EQ.0) GO TO 4
C
C                   GENERATE MINIMUM PEAKS
C
      CALL DNSPEC (NPEAK,NACRFT,DNPEAK,FOUR,IIII)
      IF (NPEAK.EQ.0) GO TO 4
C
C                   GENERATE ODD INTEGER FOR
C                   RANDOM GENERATOR
C
      NODD=(IFIX(FOUR)+IFIX(FIVE)+IFIX(SIX)+
     1IFIX(SEVEN))*IIII
      IF (NODD.EQ.0) GO TO 4
      IF (MOD(NODD,2).EQ.1) GO TO 3
      NODD=NODD+1
C
C                   LOADINGS PLACED IN RANDOM ORDER
C
    3 CALL MIXER (UPPEAK,DNPEAK,SEQUES,NPEAK,NODD)
C
C                   OBTAIN FATIGUE DAMAGE
C
      CALL ESDU (SEQUES,NPEAK,PREDIC)
      PREDIC=PREDIC*AJUST
    4 WRITE (6,9) NACRFT,FOUR,FIVE,SIX,SEVEN,PREDIC
    5 CONTINUE
C
      STOP
C
    6 FORMAT ('0',' NACRFT',T14,'FOUR',T29,'FIVE',T44,'SIX',
     1T59,'SEVEN',T73,'PREDIC',//)
    7 FORMAT (I10)
    8 FORMAT (I10,4F10.0)
    9 FORMAT ('0',I8,T15,F4.0,T29,F4.0,T43,F4.0,T60,F4.0,
     1T71,F8.7,//)
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C                         PROGRAM ESDU UTILIZES THE ESDU
C                         CUMULATIVE DAMAGE HYPOTHESIS FOR
C                         PREDICTING THE EXPENDED FATIGUE
C                         LIFE OF A-7B A/C FROM ACCELERCMETER
C                         READINGS AT THE 4,5,6, AND 7 G LEVEL
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      DIMENSION UPPEAK(5000), DNPEAK(5000), SEQUES(10001)
      WRITE (6,6)
      READ (5,7) NUM
C
      DO 5
      READ (5,8) NACRFT,FOUR,FIVE,SIX,SEVEN
      AJUST=1.0
      PREDIC=0.0
      IF (FOUR.GT.325.0) GO TO 1
      GO TO 2
    1 CALL FIX (FOUR,FIVE,SIX,SEVEN,AJUST)
C
C                         ASSURES MEMORY REQUIREMENTS SATISFIED
C
C
C                         SUBROUTINE TO GENERATE MAXIMUM G PEAKS
C                         A/C HAS EXPERIENCED
C
    2 CALL UPSPEC (NPEAK,NACRFT,FOUR,FIVE,SIX,
     1SEVEN,UPPEAK,IIII)
      IF (NPEAK.EQ.0) GO TO 4
C
C                         GENERATE MINIMUM PEAKS
C
      CALL DNSPEC (NPEAK,NACRFT,DNPEAK,FOUR,IIII)
      IF (NPEAK.EQ.0) GO TO 4
C
C                         GENERATE ODD INTEGER FOR
C                         RANDOM GENERATOR
C
      NODD=(IFIX(FOUR)+IFIX(FIVE)+IFIX(SIX)+
     1IFIX(SEVEN))*IIII
      IF (NODD.EQ.0) GO TO 4
      IF (MOD(NODD,2).EQ.1) GO TO 3
      NODD=NODD+1
C
C                         LOADINGS PLACED IN RANDOM ORDER
C
    3 CALL MIXER (UPPEAK,DNPEAK,SEQUES,NPEAK,NODD)
C
C                         OBTAIN FATIGUE DAMAGE
C
      CALL ESDU (SEQUES,NPEAK,PREDIC)
      PREDIC=PREDIC*AJUST
    4 WRITE (6,9) NACRFT,FOUR,FIVE,SIX,SEVEN,PREDIC
    5 CONTINUE
C
      STOP
C
    6 FORMAT ('0',' NACRFT',T14,'FOUR',T29,'FIVE',T44,'SIX',
     1T59,'SEVEN',T73,'PREDIC',//)
    7 FORMAT (I10)
    8 FORMAT (I10,4F10.0)
    9 FORMAT ('0',I8,T15,F4.0,T29,F4.0,T43,F4.0,T60,F4.0,
     1T71,F8.7,//)
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                    SUBROUTINE REDUCES DATA FOR COMPATIBLE
C                    MEMORY STORAGE WITH DIMENSIONED ARRAYS
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE FIX (FOUR,FIVE,SIX,SEVEN,AJUST)
      X=325.0/FOUR
      FOUR=325.0
      FIVE=X*FIVE
      SIX=X*SIX
      SEVEN=X*SEVEN
      IF (AMOD(FIVE,1.).GT.0.5) FIVE=FIVE+1.0
      IF (AMOD(SIX,1.).GT.0.5) SIX=SIX+1.0
      IF (AMOD(SEVEN,1.).GT.0.5) SEVEN=SEVEN+1.0
      FIVE=AINT(FIVE)
      SIX=AINT(SIX)
      SEVEN=AINT(SEVEN)
      AJUST=1.0/X
      RETURN
      END



CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                    SUBROUTINE GENERATES MAX G PEAKS
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


SUBROUTINE UPSPEC (NPEAK,NACRFT,FOUR,FIVE,SIX,SEVEN,
     1UPPEAK,IIII)
      DIMENSION UPPEAK(5000), NPEK(12)
C
C                    USING NORMAL DISTRIBUTICN CHOOSE NUMBER
C                    OF PEAKS IN EACH LEVEL
C
C
      DO 1 I=1,12
    1 NPEK(I)=0
C
C
C                    NEED ODD INTEGER TO INITIATE CALL
C
      NODD=IFIX(FOUR)+IFIX(FIVE)+IFIX(SIX)+IFIX(SEVEN)
      IF (NODD.EQ.0) GO TO 13
      NODD=(NACRFT/NODD)*IIII
      IF (MOD(NODD,2).EQ.1) GO TO 2
      NODD=NODD+1
    2 IX=NODD
C
C                    FIRST RANGE
C
      IF (IFIX(FOUR).EQ.0) GO TO 12
C
C                    NORMAL DISTIRBUTION RANDOM NUMBER
C                    GENERATOR
C
    3 CALL GAUSS (IX,.9764,4.01,V)
      IF (V.GT.5.62) GO TO 3
      IF (V.LT.2.399) GO TO 3
      V=FOUR*V
      NPEK(1)=IFIX(V)
C
C                    ROUND OFF TO NEAREST WHOLE NUMBER
```

```
C
      IF (AMOD(V,1.).GT.0.5) NPEK(1)=NPEK(1)+1
C
C                        SECOND RANGE
C
    4 CALL GAUSS (IX,.372,1.92,V)
      IF (V.GT.2.533) GO TO 4
      IF (V.LT.1.306) GO TO 4
      V=FOUR*V
      NPEK(2)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(2)=NPEK(2)+1
C
C                        THIRD RANGE
C
    5 CALL GAUSS (IX,.5058,1.65,V)
      IF (V.GT.2.484) GO TO 5
      IF (V.LT..8154) GO TO 5
      V=FOUR*V
      NPEK(3)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(3)=NPEK(3)+1
C
C                        FOURTH RANGE
C
    6 CALL GAUSS (IX,.314,.947,V)
      IF (V.GT.1.465) GO TO 6
      IF (V.LT..4289) GO TO 6
      V=FOUR*V
      NPEK(4)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(4)=NPEK(4)+1
C
C                        FIFTH RANGE
C
    7 CALL GAUSS (IX,.101,.6315,V)
      IF (V.GT..79815) GO TO 7
      IF (V.LT..46485) GO TO 7
      V=FOUR*V
      NPEK(5)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(5)=NPEK(5)+1
C
C                        SIXTH RANGE
C
    8 CALL GAUSS (IX,.0282,.417,V)
      IF (V.GT..46353) GO TO 8
      IF (V.LT..37047) GO TO 8
      V=FOUR*V
      NPEK(6)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(6)=NPEK(6)+1
C
C                        SEVENTH RANGE
C
    9 CALL GAUSS (IX,.0195,.324,V)
      IF (V.GT..3562) GO TO 9
      IF (V.LT..2918) GO TO 9
      V=FOUR*V
      NPEK(7)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(7)=NPEK(7)+1
      IF (FOUR.LT.1.1) NPEK(7)=1
C
C                        EIGHTH RANGE
C
   10 CALL GAUSS (IX,.0111,.263,V)
      IF (V.GT..281) GO TO 10
      IF (V.LT..2447) GO TO 10
      V=FOUR*V
      NPEK(8)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(8)=NPEK(8)+1
      IF (FOUR.LT.2.1) NPEK(8)=1
C
C                        NINTH RANGE
C
   11 CALL GAUSS (IX,.0175,.1505,V)
      IF (V.GT..1794) GO TO 11
```

```
      IF (V.LT..1216) GO TO 11`
      V=FOUR*V
      NPEK(9)=IFIX(V)
      IF (AMOD(V,1.).GT.0.5) NPEK(9)=NPEK(9)+1
   12 NPEK(10)=IFIX(FIVE)
      NPEK(11)=IFIX(SIX)
      NPEK(12)=IFIX(SEVEN)
C
C                           FILL VECTOR WITH LOADINGS
C
   13 CALL HILOAD (NPEK,UPPEAK,NPEAK)
      RETURN
C
      END




CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                           SUBROUTINE ASSIGNS REPRESENTATIVE
C                           G LOADS TO EACH RANGE
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE HILOAD (NPEK,UPPEAK,NPEAK)
      DIMENSION NPEK(12), UPPEAK(5000)
      NPEAK=0
C
      DO 1 I=1,12
    1 NPEAK=NPEK(I)+NPEAK
C
      IF (NPEAK.EQ.0) GO TO 27
      NDUMMY=0
      NTOP=0
      IF (NPEK(1).EQ.0) GO TO 3
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(1)-1
C
      DC 2 I=NDUMMY,NTOP
    2 UPPEAK(I)=2.25
C
    3 IF (NPEK(2).EQ.0) GO TO 5
      NDUMMY=NPEK(1)+1
      NTOP=NDUMMY+NPEK(2)-1
C
      DO 4 I=NDUMMY,NTOP
    4 UPPEAK(I)=2.7
C
    5 IF (NPEK(3).EQ.0) GO TO 7
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(3)-1
C
      DO 6 I=NDUMMY,NTOP
    6 UPPEAK(I)=3.1
C
    7 IF (NPEK(4).EQ.0) GO TO 9
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(4)-1
C
      DO 8 I=NDUMMY,NTOP
    8 UPPEAK(I)=3.45
C
    9 IF (NPEK(5).EQ.0) GO TO 11
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(5)-1
C
      DO 10 I=NDUMMY,NTOP
   10 UPPEAK(I)=3.75
```

```
C
   11 IF (NPEK(6).EQ.0) GO TO 13
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(6)-1
C
      DO 12 I=NDUMMY,NTOP
   12 UPPEAK(I)=4.05
C
   13 IF (NPEK(7).EQ.0) GO TO 15
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(7)-1
C
      DO 14 I=NDUMMY,NTOP
   14 UPPEAK(I)=4.35
C
   15 IF (NPEK(8).EQ.0) GO TO 17
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(8)-1
C
      DO 16 I=NDUMMY,NTOP
   16 UPPEAK(I)=4.65
C
   17 IF (NPEK(9).EQ.0) GO TO 19
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(9)-1
C
      DO 18
   18 UPPEAK(I)=4.9
C
   19 IF (NPEK(10).EQ.0) GO TO 21
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(10)-1
C
      DO 20 I=NDUMMY,NTOP
   20 UPPEAK(I)=5.75
C
   21 IF (NPEK(11).EQ.0) GO TO 23
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(11)-1
C
      DO 22 I=NDUMMY,NTOP
   22 UPPEAK(I)=6.6
C
   23 IF (NPEK(12).EQ.0) GO TO 25
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(12)-1
C
      DO 24 I=NDUMMY,NTOP
   24 UPPEAK(I)=7.8
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                      SUBROUTINE GENERATES MINIMUM PEAKS
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE DNSPEC (NPEAK,NACRFT,DNPEAK,FOUR,IIII)
      DIMENSION DNPEAK(5000), NPEK(5)
C
      DO 1 I=1,5
    1 NPEK(I)=0
C
      IF (IFIX(FOUR).EQ.0) GO TO 7
      NODD=(NACRFT/FOUR)*IIII
      IF (MOD(NODD,2).EQ.1) GO TO 2
      NODD=NODD+1
    2 IX=NODD
C
C                      FIRST RANGE
C
    3 CALL GAUSS (IX,1.93099,7.1554,V)
      IF (V.GT.10.342) GO TO 3
      IF (V.LT.3.9692) GO TO 3
      V=FOUR*V
      NPEK(1)=IFIX(V)
      IF (AMOD(V,1.).GT..5) NPEK(1)=NPEK(1)+1
C
C                      SECOND RANGE
C
    4 CALL GAUSS (IX,.8563,1.6728,V)
      IF (V.GT.3.0857) GO TO 4
      IF (V.LT..2599) GO TO 4
      V=FOUR*V
      NPEK(2)=IFIX(V)
      IF (AMOD(V,1.).GT..5) NPEK(2)=NPEK(2)+1
C
C                      THIRD RANGE
C
    5 CALL GAUSS (IX,.18698,.35888,V)
      IF (V.GT..6674) GO TO 5
      IF (V.LT..050363) GO TO 5
      V=FOUR*V
      NPEK(3)=IFIX(V)
      IF (AMOD(V,1.).GT..5) NPEK(3)=NPEK(3)+1
C
C                      FOURTH RANGE
C
    6 CALL GAUSS (IX,.05951,.10252,V)
      IF (V.GT..200712) GO TO 6
      IF (V.LT..00433) GO TO 6
      V=FOUR*V
      NPEK(4)=IFIX(V)
      IF (AMOD(V,1.).GT..5) NPEK(4)=NPEK(4)+1
C
C                      FIFTH RANGE
C
    7 NPEK(5)=NPEAK-(NPEK(1)+NPEK(2)+NPEK(3)+NPEK(4))
      CALL LOLOAD (NPEK,DNPEAK)
      RETURN
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                                SUBROUTINE ASSIGNS G LOADINGS TO
C                                MINIMUM PEAK RANGES
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE LOLOAD (NPEK,DNPEAK)
      DIMENSION NPEK(5), DNPEAK(5000)
      NPEAK=0
C
      DO 1 I=1,5
    1 NPEAK=NPEK(I)+NPEAK
C
      NDUMMY=0
      NTOP=0
      IF (NPEK(1).EQ.0) GO TO 3
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(1)-1
C
      DO 2 I=NDUMMY,NTOP
    2 DNPEAK(I)=.538
C
    3 IF (NPEK(2).EQ.0) GO TO 5
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(2)-1
C
      DO 4 I=NDUMMY,NTOP
    4 DNPEAK(I)=.346
C
    5 IF (NPEK(3).EQ.0) GO TO 7
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(3)-1
C
      DO 6 I=NDUMMY,NTOP
    6 DNPEAK(I)=.148
C
    7 IF (NPEK(4).EQ.0) GO TO 9
      NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(4)-1
C
      DO 8 I=NDUMMY,NTOP
    8 DNPEAK(I)=-.054
C
    9 NDUMMY=NTOP+1
      NTOP=NDUMMY+NPEK(5)-1
C
      DO 10 I=NDUMMY,NTOP
   10 DNPEAK(I)=1.0
C
      IF (NPEAK.NE.NTOP) GO TO 11
      GO TO 12
   11 WRITE (6,13)
      NPEAK=0
   12 RETURN
C
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                    SUBROUTINE GENERATES UNIFORMLY
C                    DISTRIBUTED RANDOM NUMBER
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE RANDU (IX,IY,YFL)
      IY=IX*65539
      IF (IY) 1,2,2
    1 IY=IY+2147483647+1
    2 YFL=IY
      YFL=YFL*.4656613E-9
      RETURN
      END




CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                    SUBROUTINE GENERATES NORMALLY
C                    DISTRIBUTED RANDOM NUMBER
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE GAUSS (IX,S,AM,V)
      A=0.0
C
      DO 1 I=1,12
C
C                    UNIFORMLY DISTRIBUTED RANDOM
C                    NUMBER GENERATOR
C
      CALL RANDU (IX,IY,Y)
      IX=IY
    1 A=A+Y
C
      V=(A-6.0)*S+AM
      RETURN
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                       SUBROUTINE ARRANGES PEAKS IN
C                       A RANDOM ORDER
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE MIXER (UPPEAK,DNPEAK,SEQUES,NPEAK,NODD)
      DIMENSION UPPEAK(5000), DNPEAK(5000), SEQUES(10001),
     1M(5000)
      NOTEVN=NODD
      CALL SHUFLE (M,NPEAK,NOTEVN)
C
C                       RANDOMIZE INDEX OF LOADS
C
C
      DO 1 I=1,NPEAK
    1 SEQUES(2*I-1)=DNPEAK(M(I))
C
      NOTEVN=NODD+8
      CALL SHUFLE (M,NPEAK,NOTEVN)
C
      DO 2 I=1,NPEAK
    2 SEQUES(2*I)=UPPEAK(M(I))
C
      SEQUES(2*NPEAK+1)=1.0
      RETURN
      END


CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                       SUBROUTINE  SHUFFLES INDEXES OF M
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE SHUFLE (M,NPEAK,NOTEVN)
      DIMENSION M(NPEAK)
      CALL OVFLOW
C
      DO 1 I=1,NPEAK
      M(I)=I
    1 CONTINUE
C
      K=NPEAK
    2 CALL RANDOM (NOTEVN,U,1)
      I=K*U+1
      L=M(I)
      M(I)=M(K)
      M(K)=L
      K=K-1
      IF (K.GT.1) GO TO 2
      L=M(NPEAK)
      M(NPEAK)=M(2)
      M(2)=L
      RETURN
      END
```

```
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C
C                         SUBROUTINE CALCULATES FATIGUE LIFE
C                         BY ESDU METHOD
C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC


      SUBROUTINE ESDU (SEQUES,NPEAK,PREDIC)
      DIMENSION SEQUES(10001)
      CORECT=0.0
      TK=3.5
      BCT=3.6
      SLITLE=5.5
      FT=77.0
      FP=69.0
      SMELCA=10.0
      PREDIC=0.0
      NMOST=2*NPEAK+1
C
      DO 1 I=1,NMOST
C
C                         CONVERT TO STRESSES IN KSI
C
    1 SEQUES(I)=3.5143*SEQUES(I)
C
      NMOST=NMOST-1
C
C                         COMPUTE STRESS MEAN ACTUAL PROGRAM
C
C
C
      DO 16
      ORRECT=0.0
      SM=(SEQUES(I+1)+SEQUES(I))/2.0
      SA=ABS(SEQUES(I+1)-SEQUES(I))/2.0
C
C                         CONVERT TO ACTUAL STRESS
C
      SMEL=TK*SM
      SAEL=TK*SA
      SMAXEL=SMEL+SAEL
      SMINEL=SMEL-SAEL
C
C                         CHECKING FOR TYPE 4
C
      IF (SAEL.GT.FP) GO TO 2
      GO TO 3
    2 IF (SEQUES(I+1).GT.SEQUES(I)) GO TO 4
      IF (SEQUES(I+1).LT.SEQUES(I)) GO TO 6
    3 IF ((CORECT+SMAXEL).GT.FP) GO TO 4
      GO TO 5
C
C                         TYPE 3
C
    4 CORECT=FP-SMAXEL
      GO TO 7
    5 IF ((CORECT+SMINEL).LT.(-1.*FP)) GO TO 6
      GO TO 7
C
C                         TYPE 2
C
    6 CORECT=-1.0*(SMINEL+FP)
    7 SMACP=SMEL+CORECT
      SMAXEC=SMELCA+SAEL
C
C                         DOWN LOADS NO CONTRIBUTION
C                         TO DAMAGE
C
      IF (SEQUES(I+1).LT.SEQUES(I)) GO TO 16
C
C                         ADJUST FOR S N DATA
```

```
C
      IF  (SMAXEC.GT.FP) ORRECT=FP-SMAXEC
      SMACA=ORRECT+SMELCA
      BOTCAL=1.0-(SMACP/FT)**2
      TOPCAL=1.0-(SMACA/FT)**2
      IF  (SMACA.LT.0.0)  TOPCAL=1.0+(SMACA/FT)**2
      IF  (SMACP.LT.0.0)  BOTCAL=1.0+(SMACP/FT)**2
      SAELCA=(TOPCAL/BOTCAL)*SAEL
      SAELCA=SAELCA/TK
      IF  (SAELCA.LT.BOT) GO TO 16
C
C                       S N DATA
C
C
      CYCLES=9.8596-.5981*SAELCA+.014207*(SAELCA**2)
      CYCLES=10**CYCLES
      PREDIC=1.0/CYCLES+PREDIC
   16 CONTINUE
C
      RETURN
      END
```

APPENDIX B

Figures

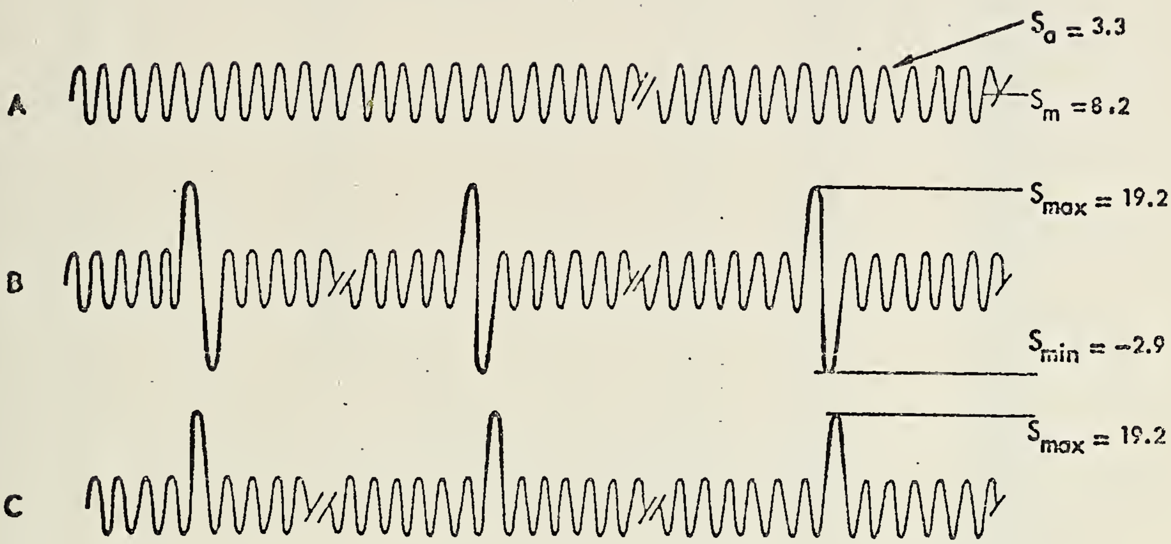

FIG. 1a THREE DIFFERENT LOADING SPECTRUMS
APPLIED TO SPECIMENS.

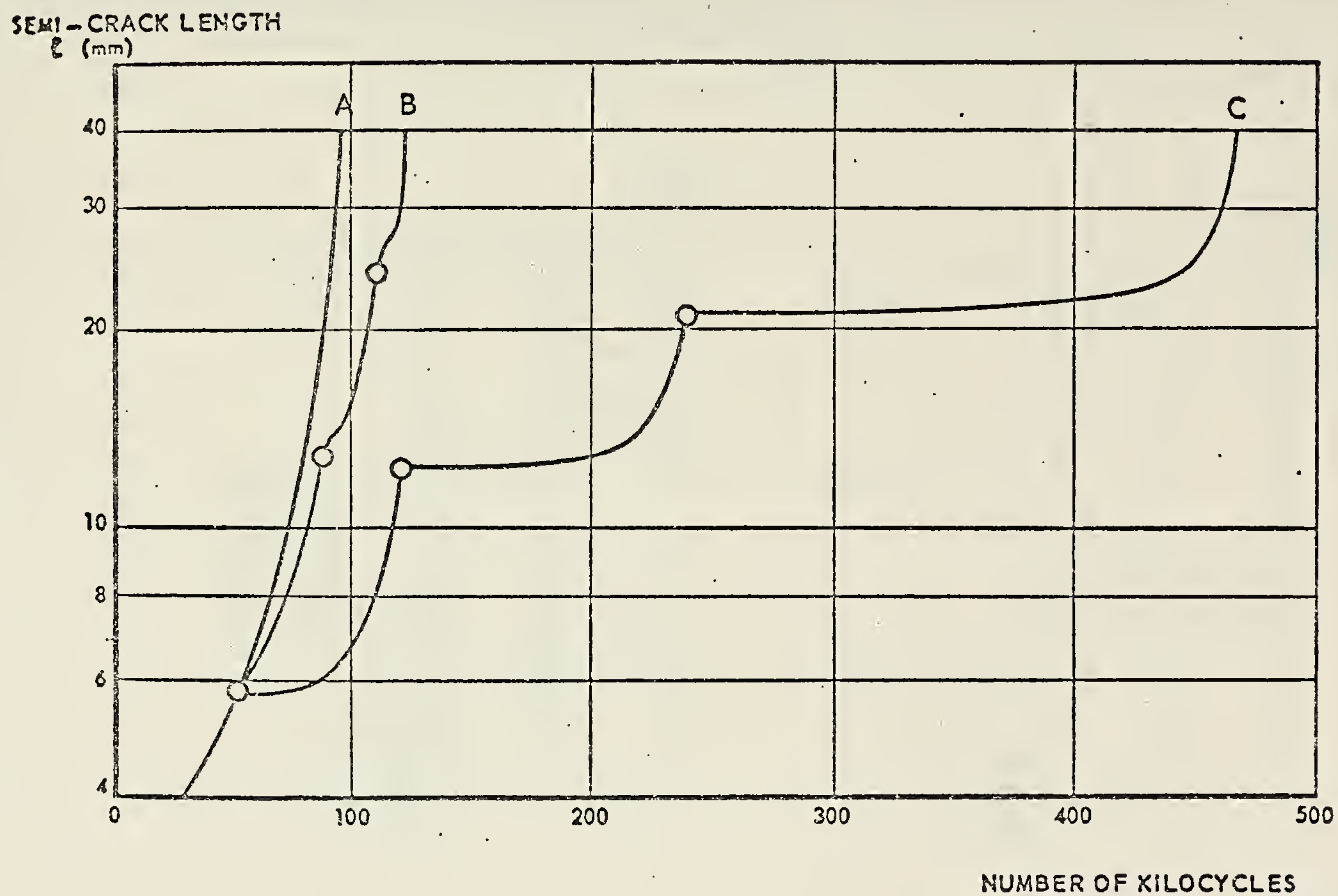

FIG. 1b THE DELAYING EFFECT OF PEAK LOADS ON CRACK PROPAGATION IN 2024-T3 ALCLAD SHEET SPECIMENS, WIDTH 160 mm, THICKNESS 2 mm (REF. 27).

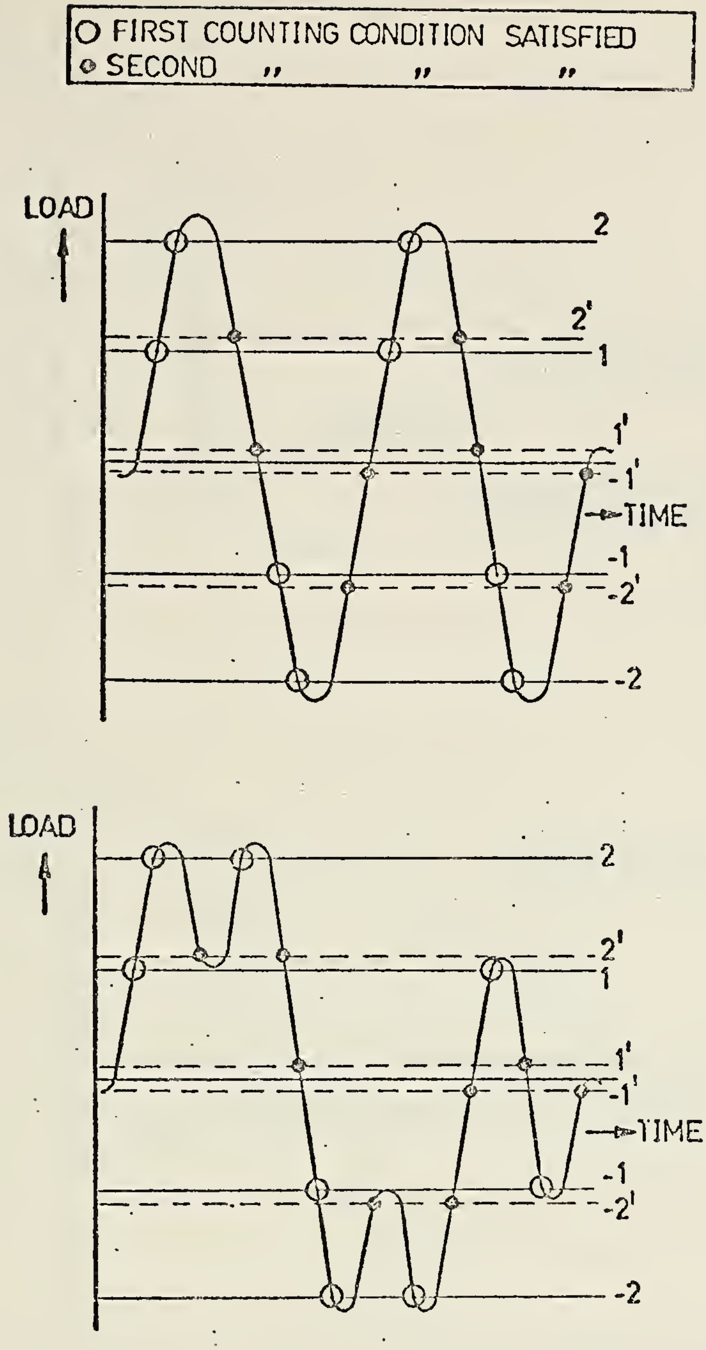

FIG. 2 RESTRICTED LEVEL-CROSSING METHOD.

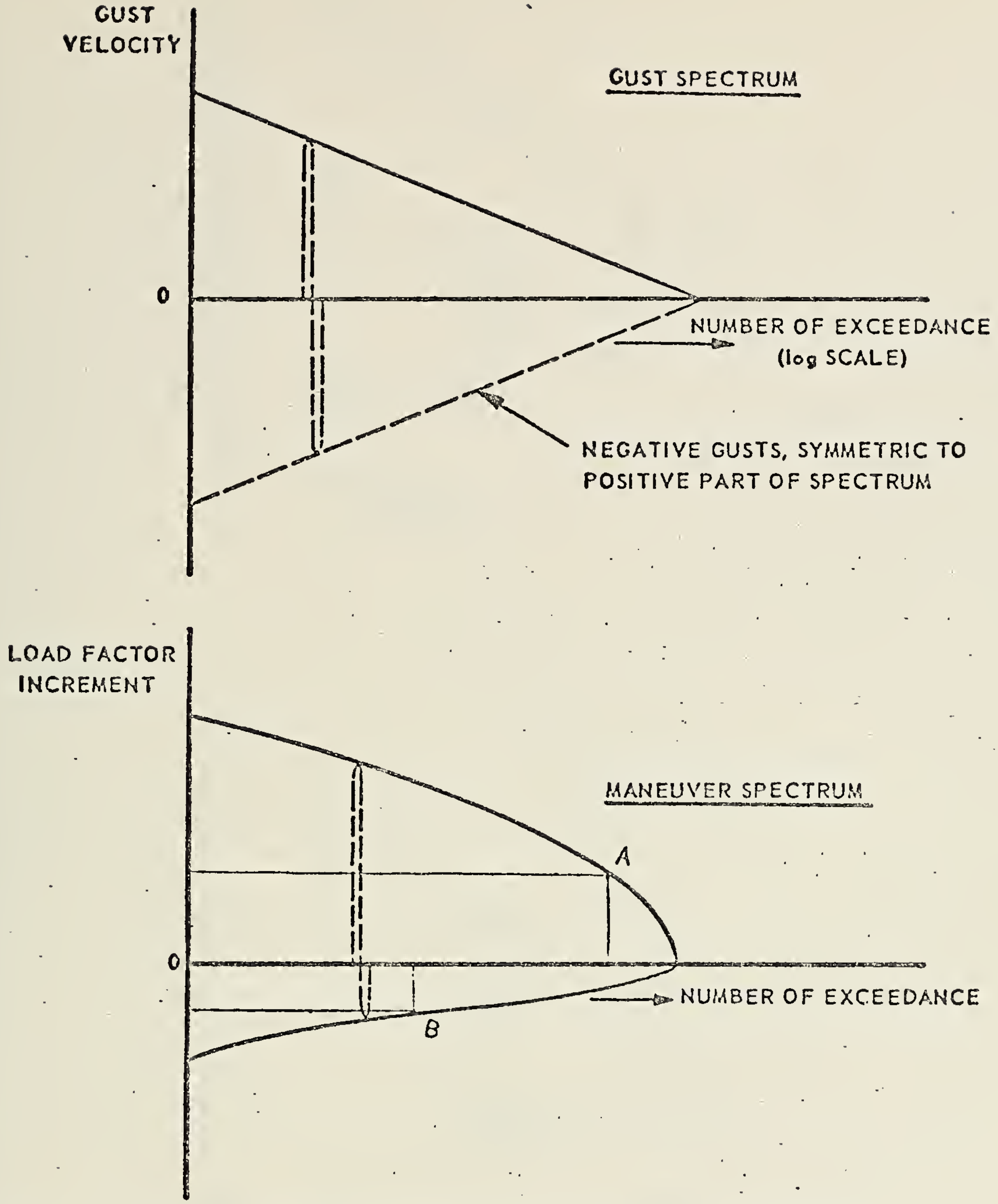

FIG. 3 DIFFERENCES BETWEEN GUST AND MANEUVER SPECTRA

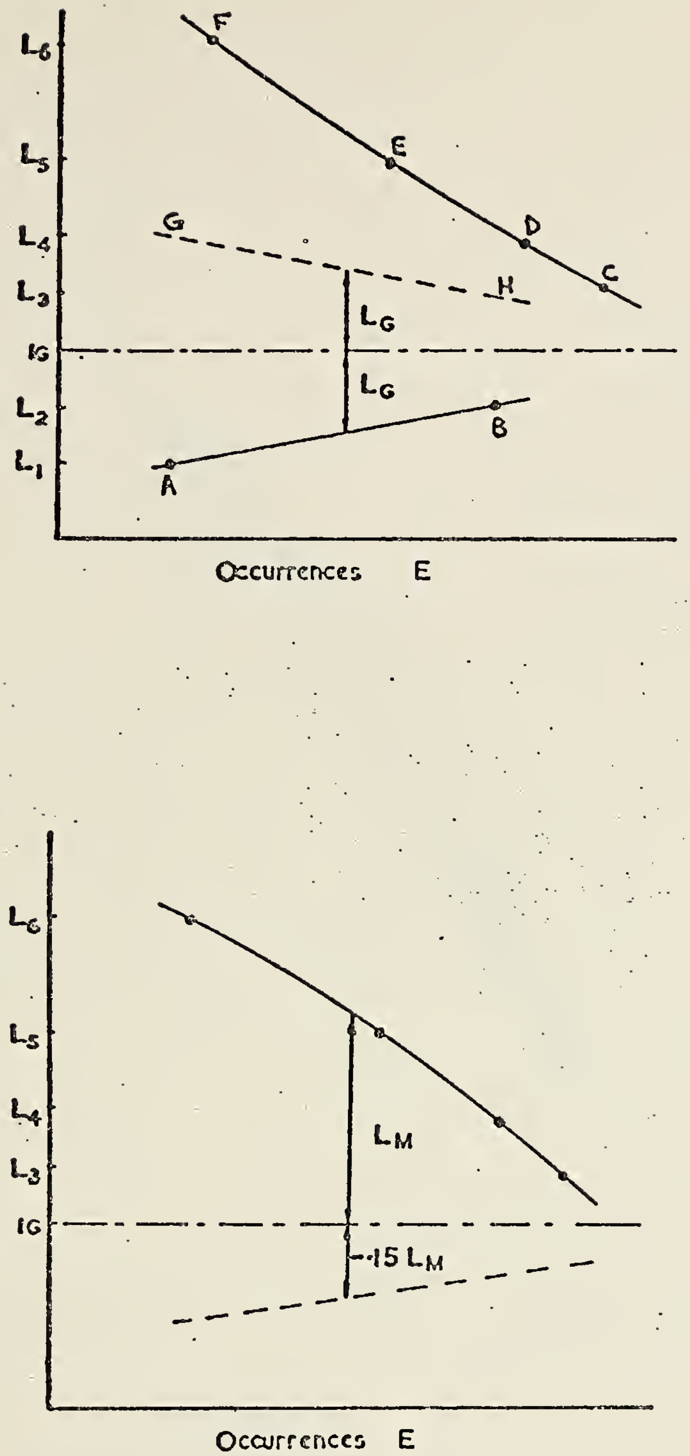

FIG. 4 GUST LOADING SPECTRUM (TOP) AND MANEUVER SPECTRUM (BOTTOM).

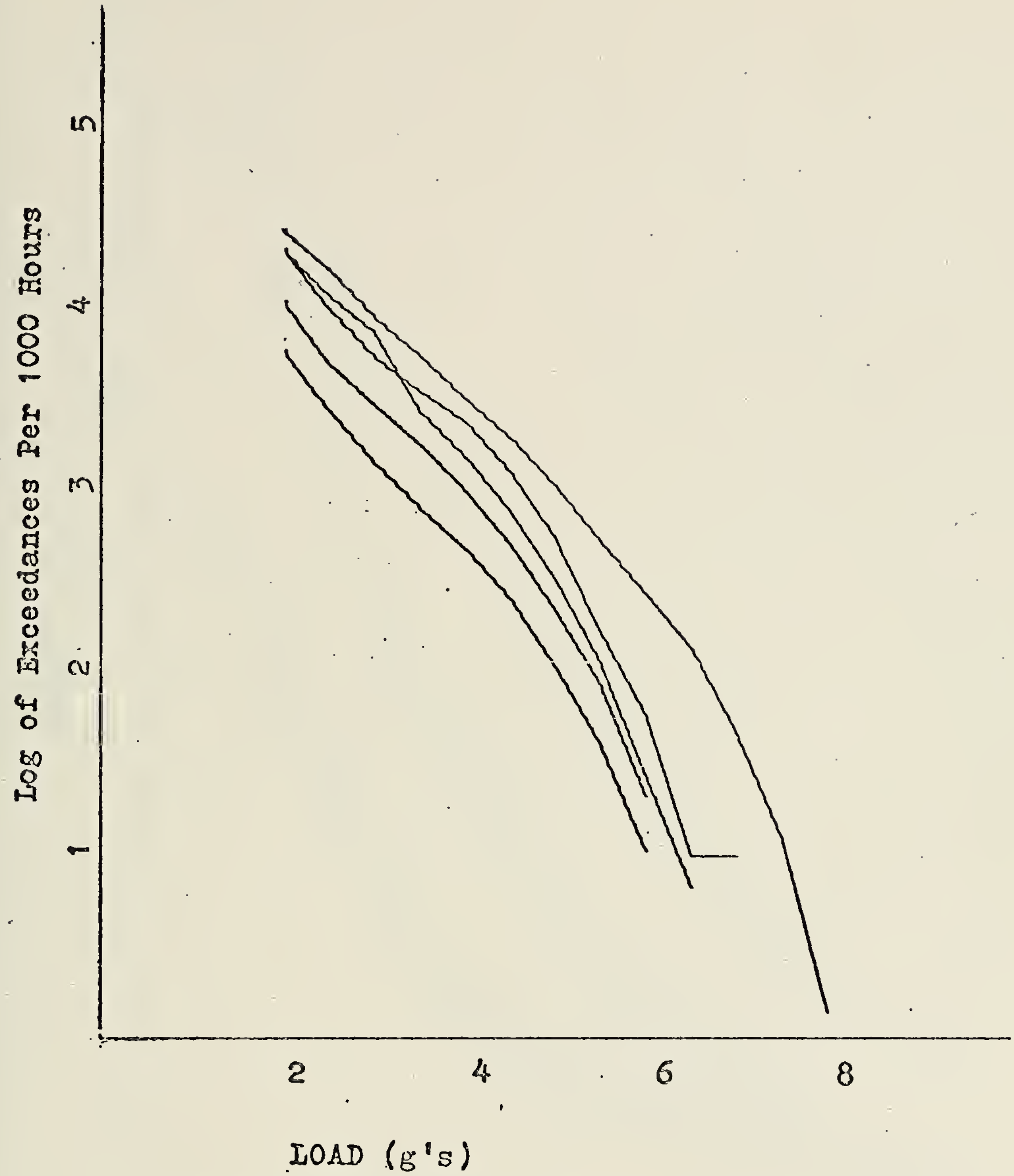

FIG. 5 A-7B, A-6A, A-4E, F-4B, AND F-8E
EXCEEDANCES PER 1000 HOURS.

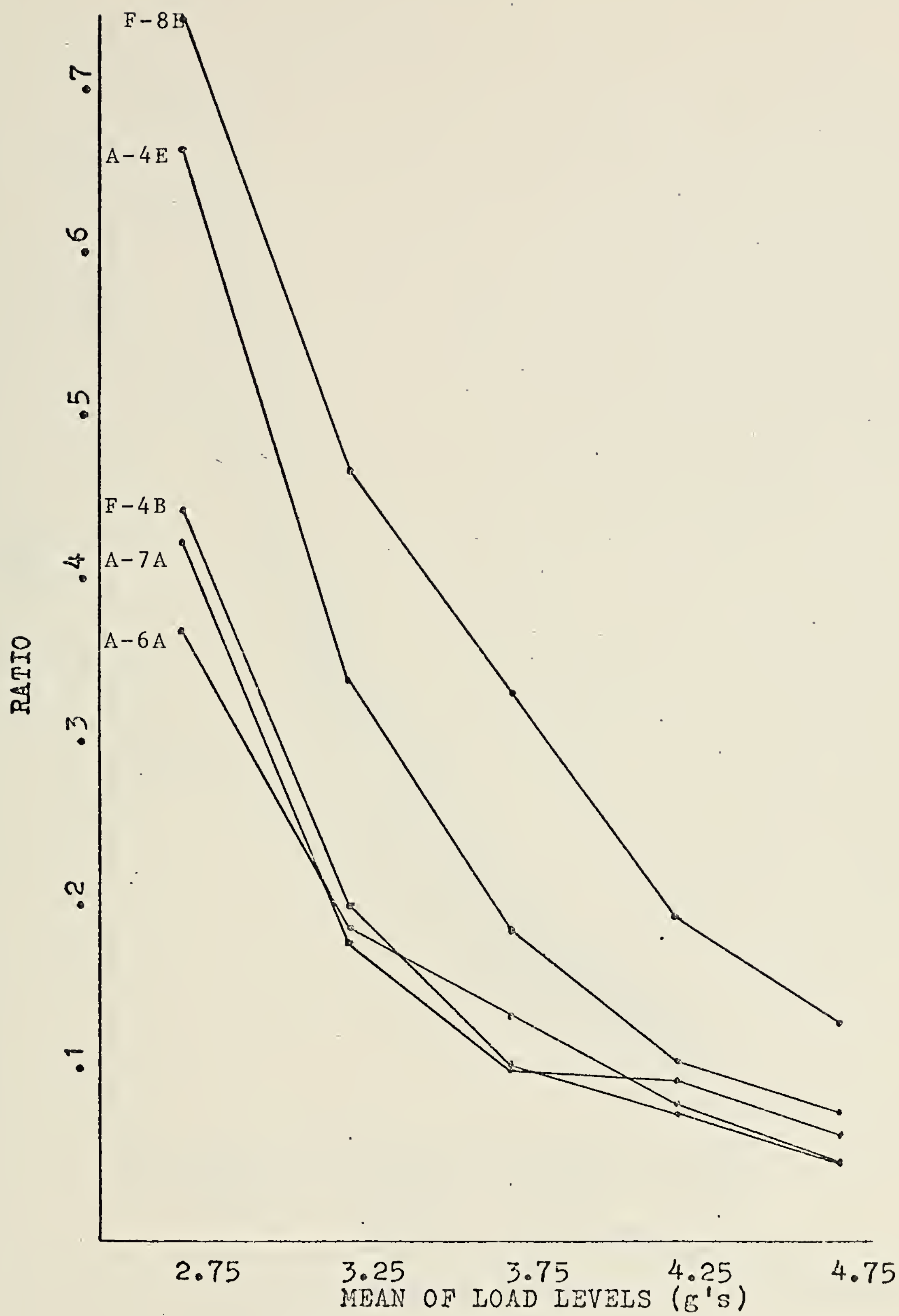

FIG. 6 RATIO OF 2-2.5 g LOADS TO OTHER LEVELS.

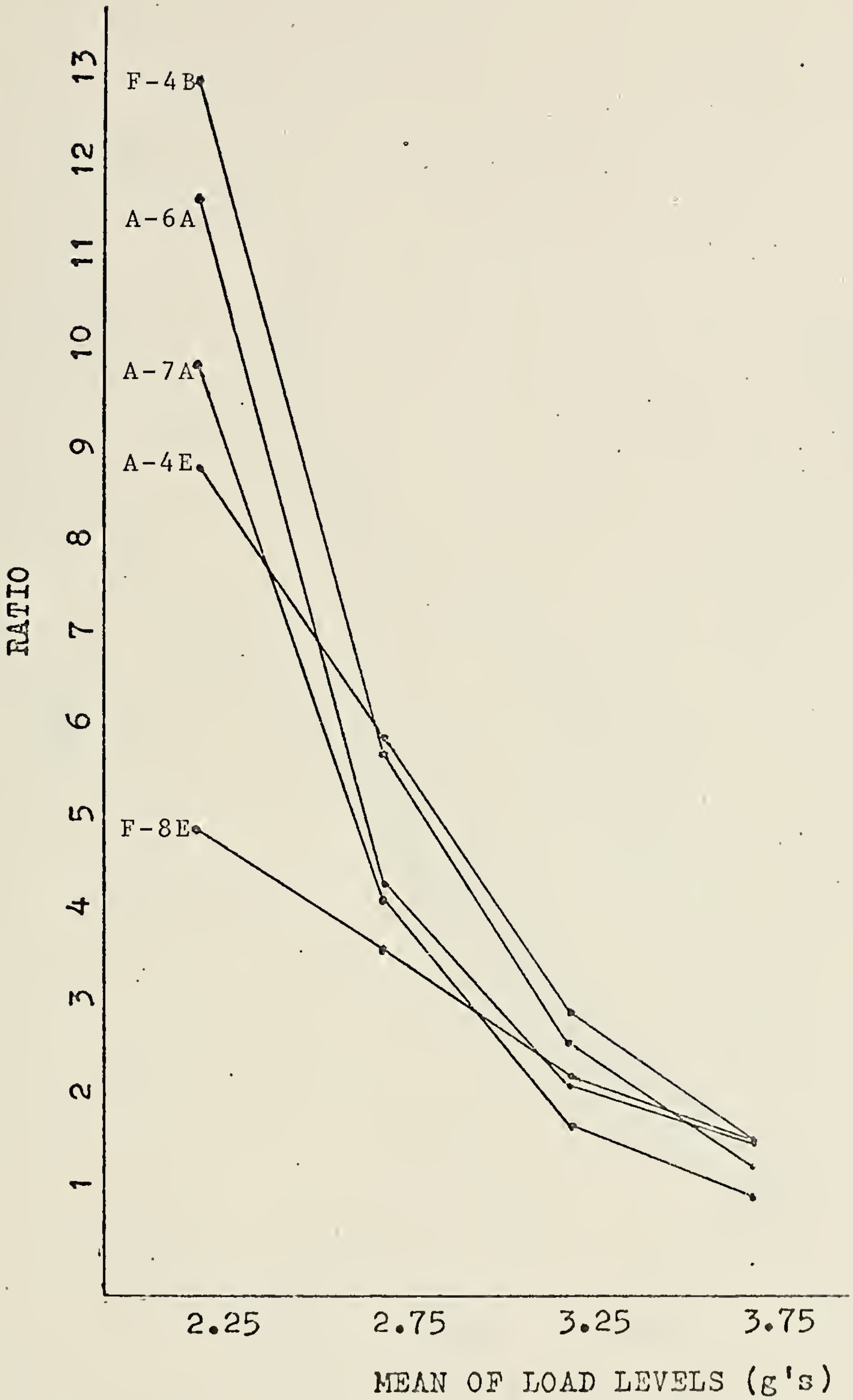

FIG. 7 RATIO OF LOAD TO 4-4.5 g's FOR FIVE A/C.

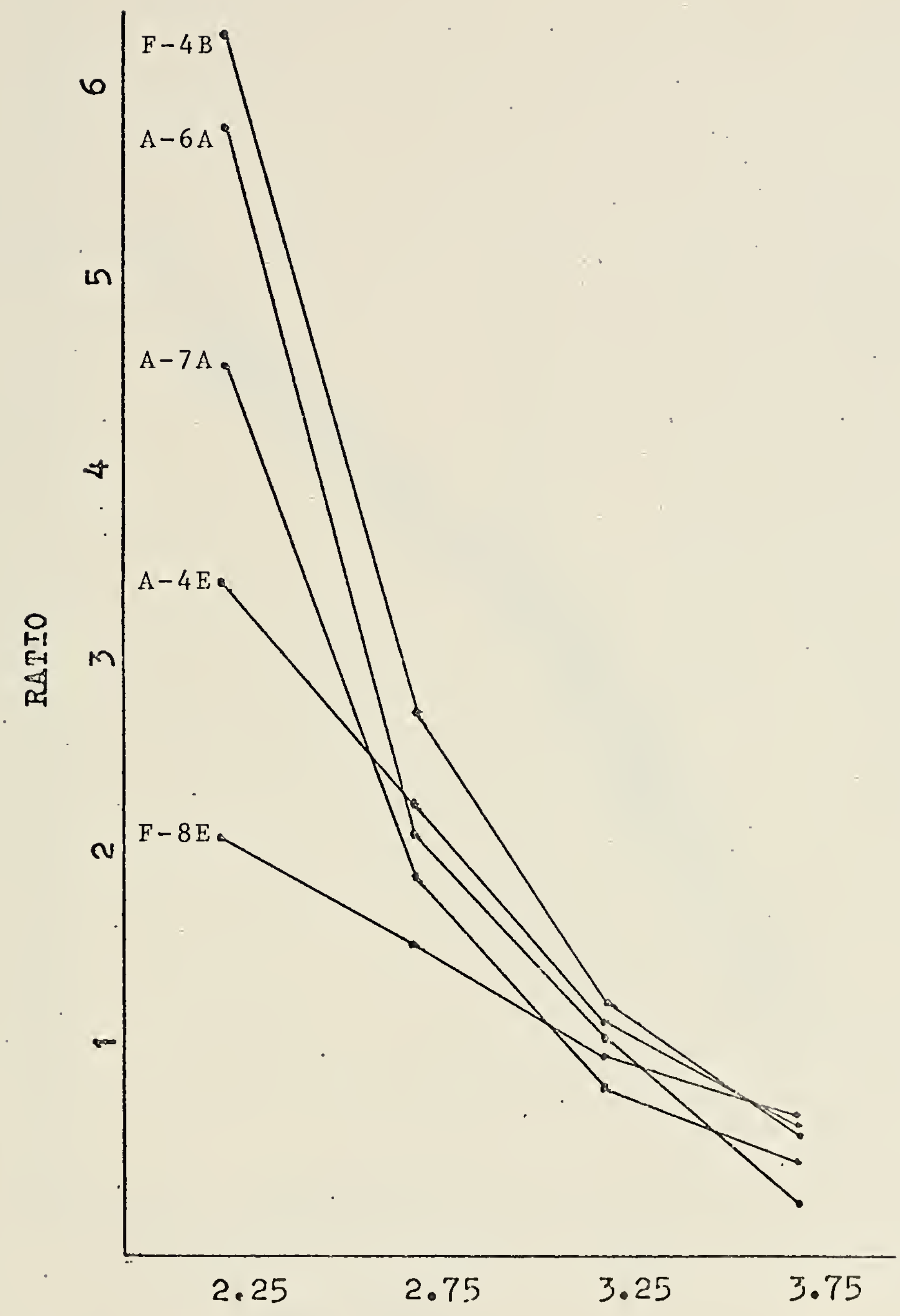

FIG. 8 RATIO OF LOADS TO 4 g EXCEEDANCES.

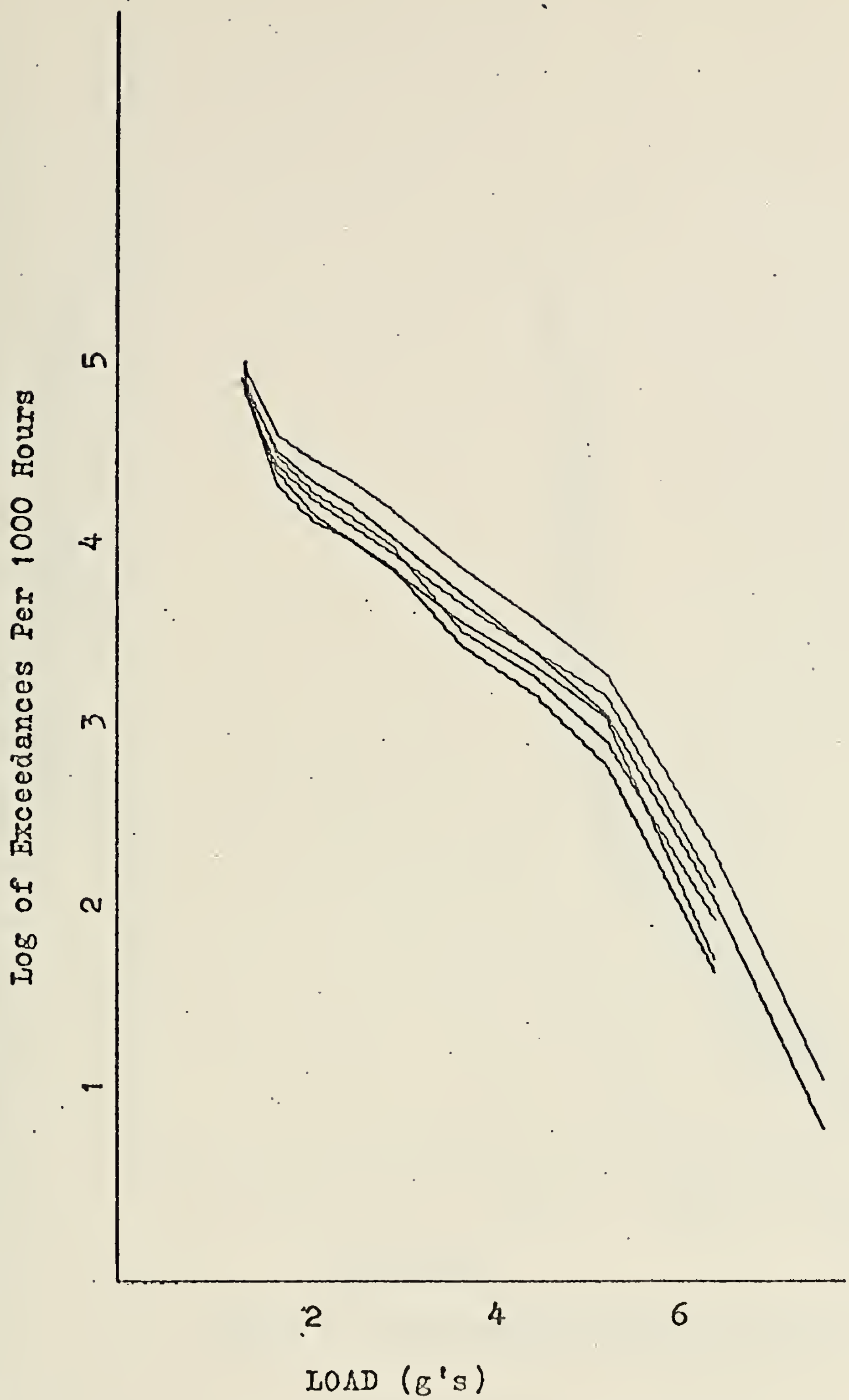

FIG. 9 A-7B EXCEEDANCES PER 1000 HOURS.

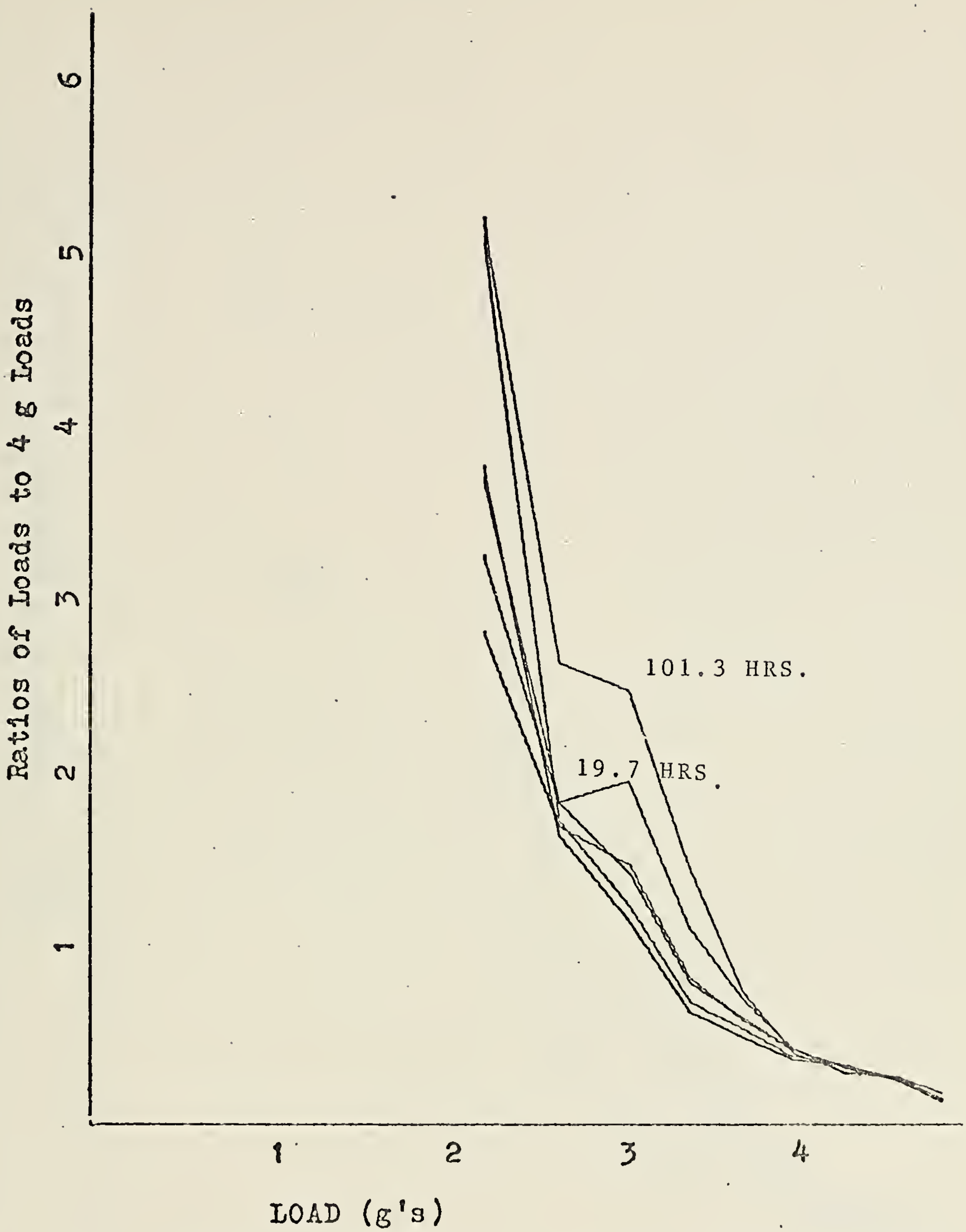

FIG. 10 A-7B RATIO OF LOADS TO 4-5 g LOADS.

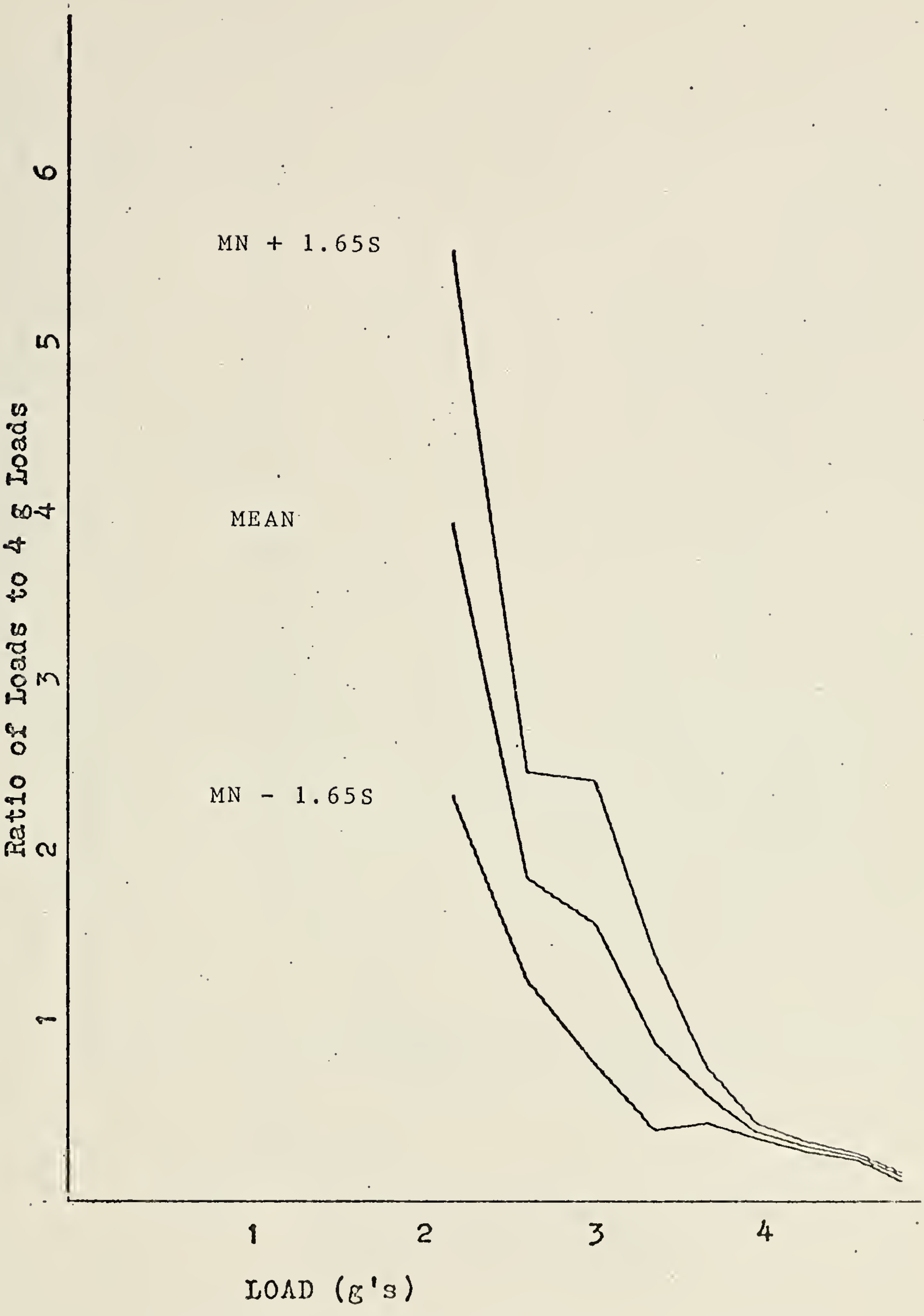

FIG.11 A-7B PROPOSED RATIO OF LOADS TO 4-5 g LOADS.

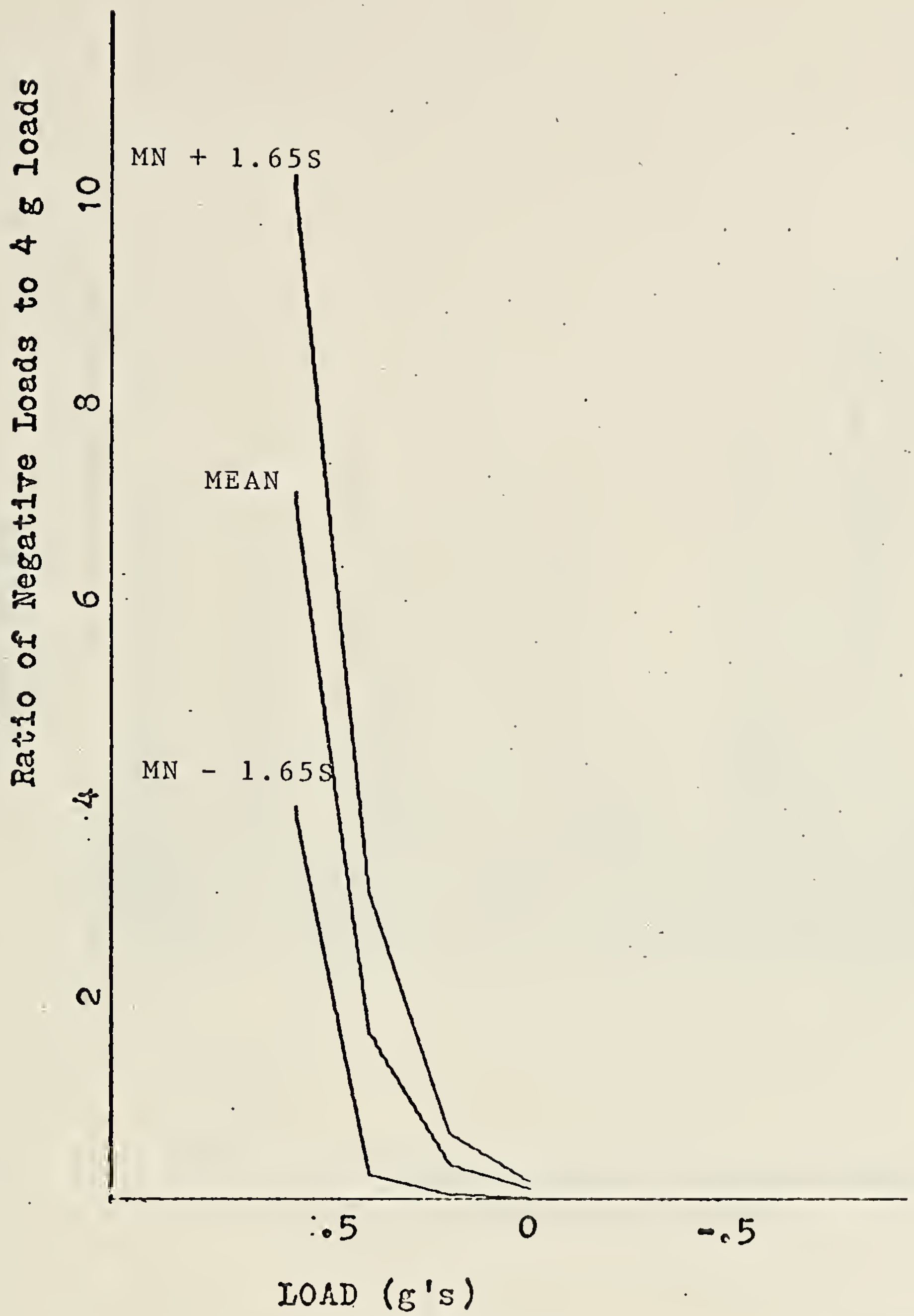

FIG. 12 A-7B PROPOSED RATIO OF NEGATIVE LOADS TO 4-5 g LOADS.

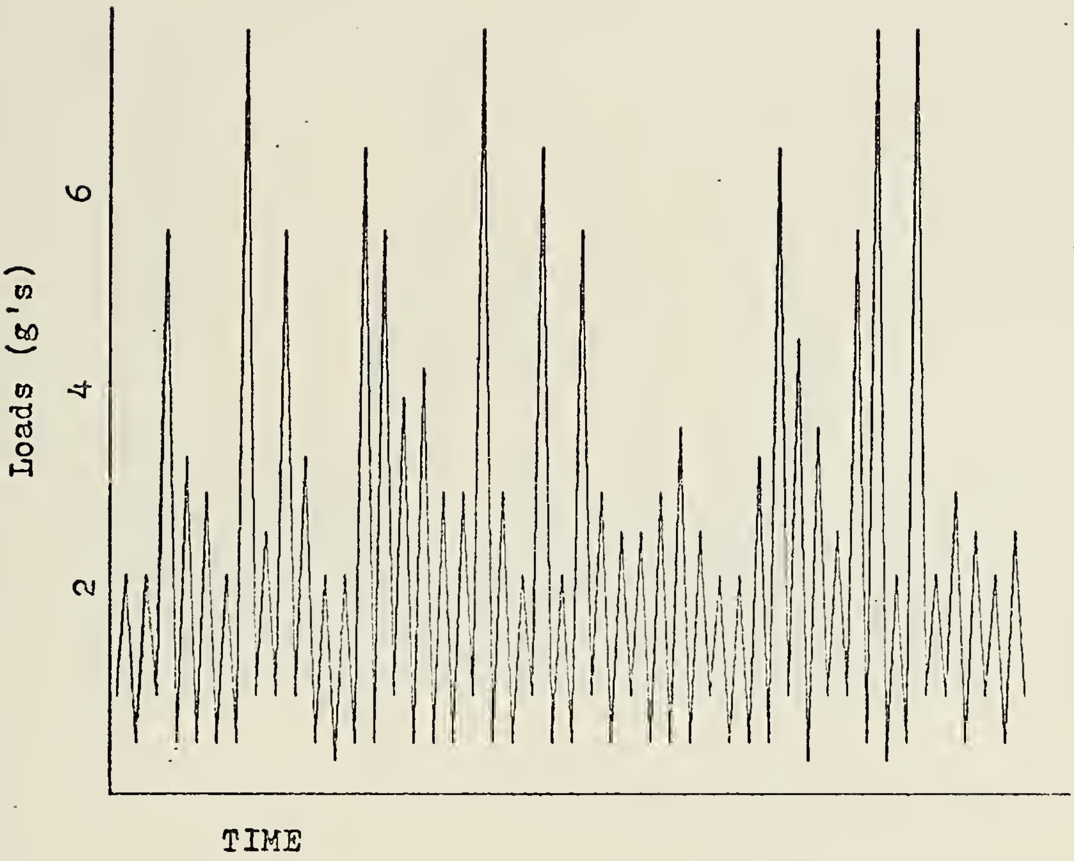

FIG.13 A-7B LOAD SEQUENCE BASED ON ACCELEROMETER READINGS OF 3,5,3, AND 4.

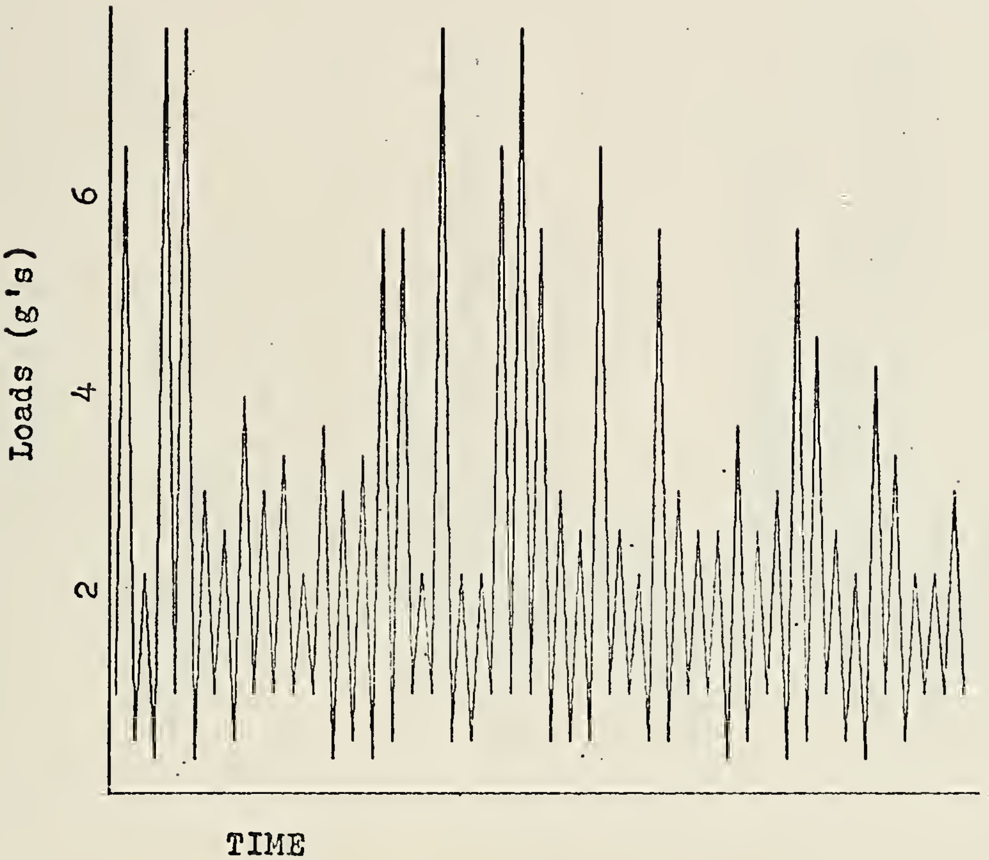

FIG. 14 A-7B LOAD SEQUENCE BASED ON ACCELEROMETER READINGS OF 3, 5, 3, AND 4.

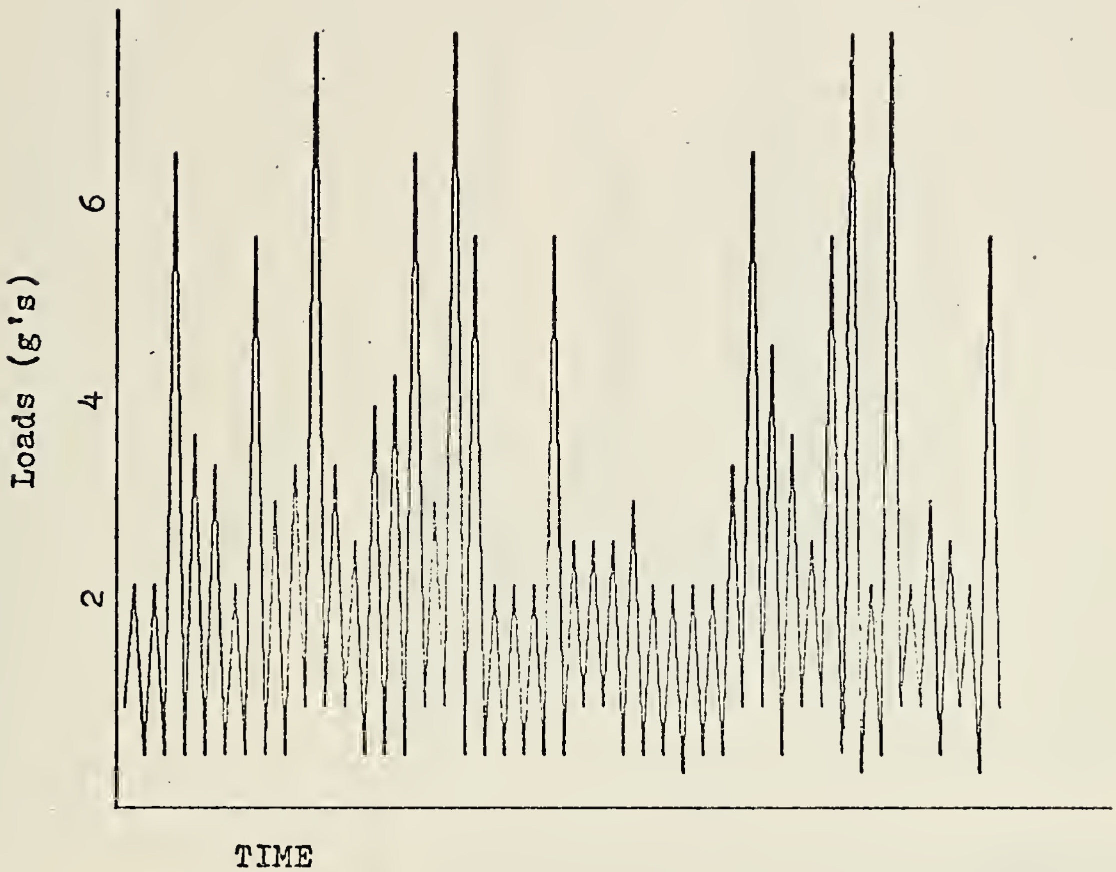

FIG.15 A-7B LOAD SEQUENCE BASED ON ACCELEROMETER READINGS OF 3,5,3, AND 4.

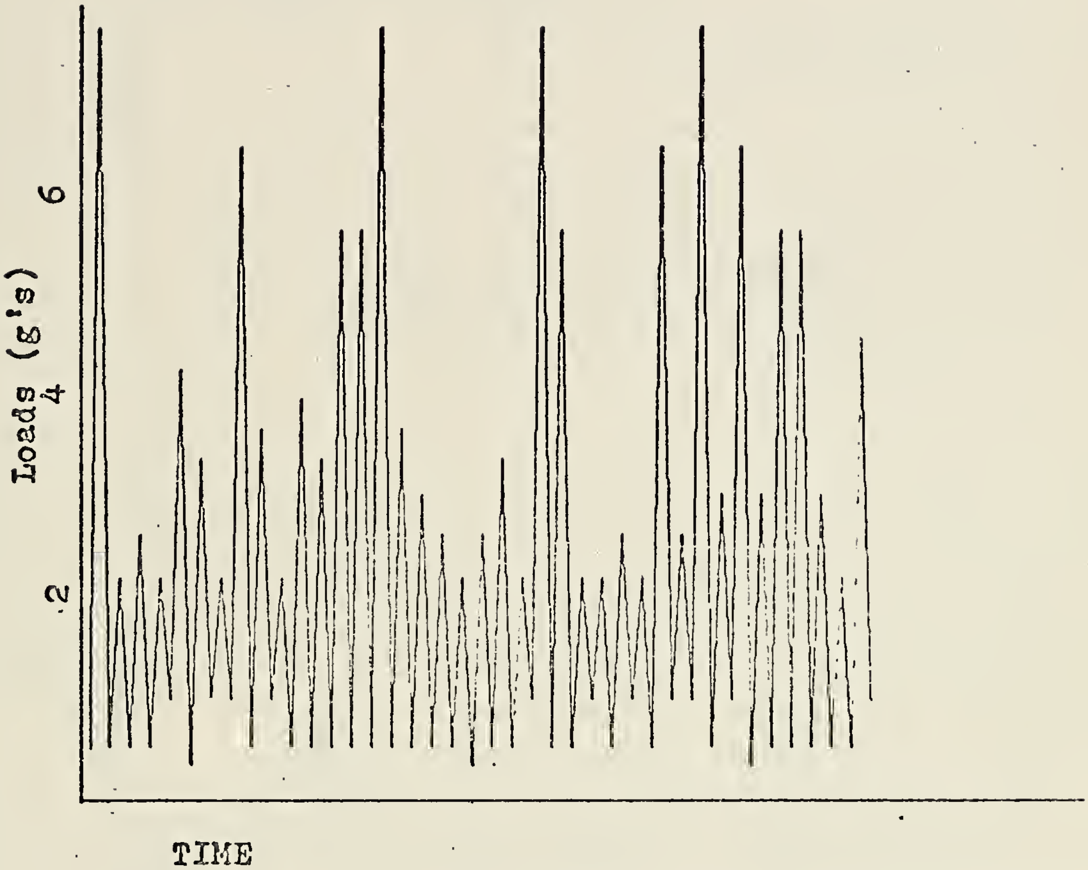

FIG. 16 A-7B LOAD SEQUENCE BASED ON ACCELEROMETER READINGS OF 3, 5, 3, AND 4.

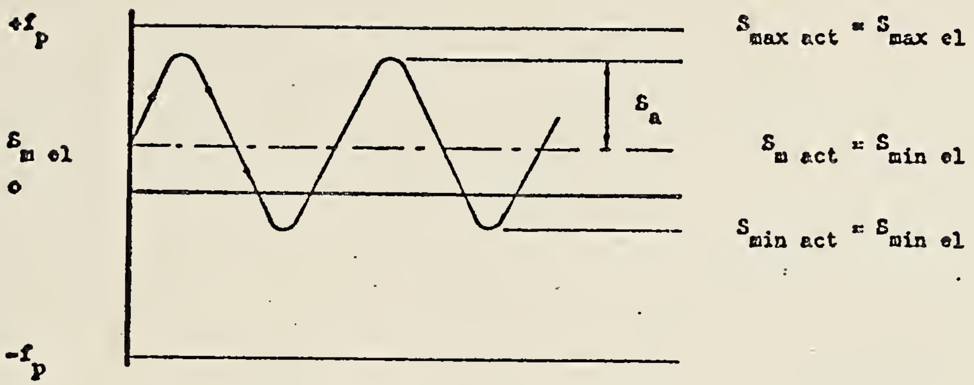

FIG.17 TYPE 1 LOAD CYCLES. SMAXEL
LESS THAN FP.

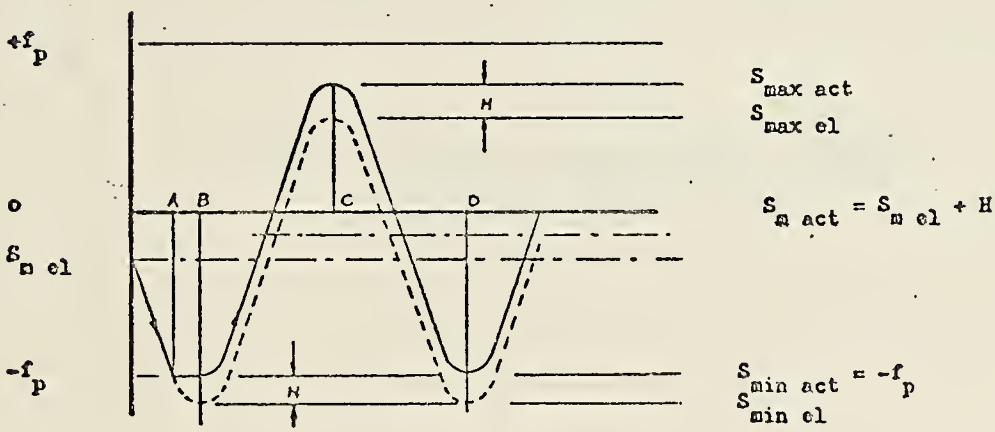

FIG.18 TYPE 2 LOAD CYCLES. SMINEL
LESS THAN -FP.

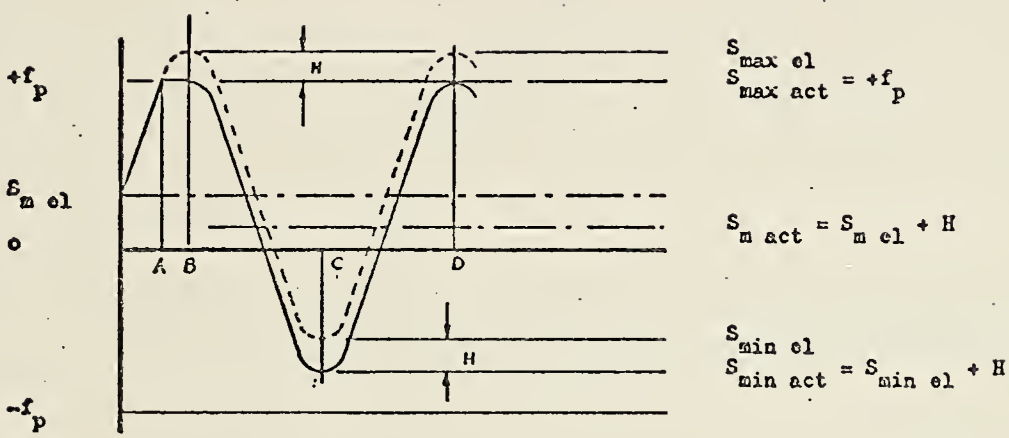

FIG. 19 TYPE 3 LOAD CYCLES. SMAXEL GREATER THAN FP.

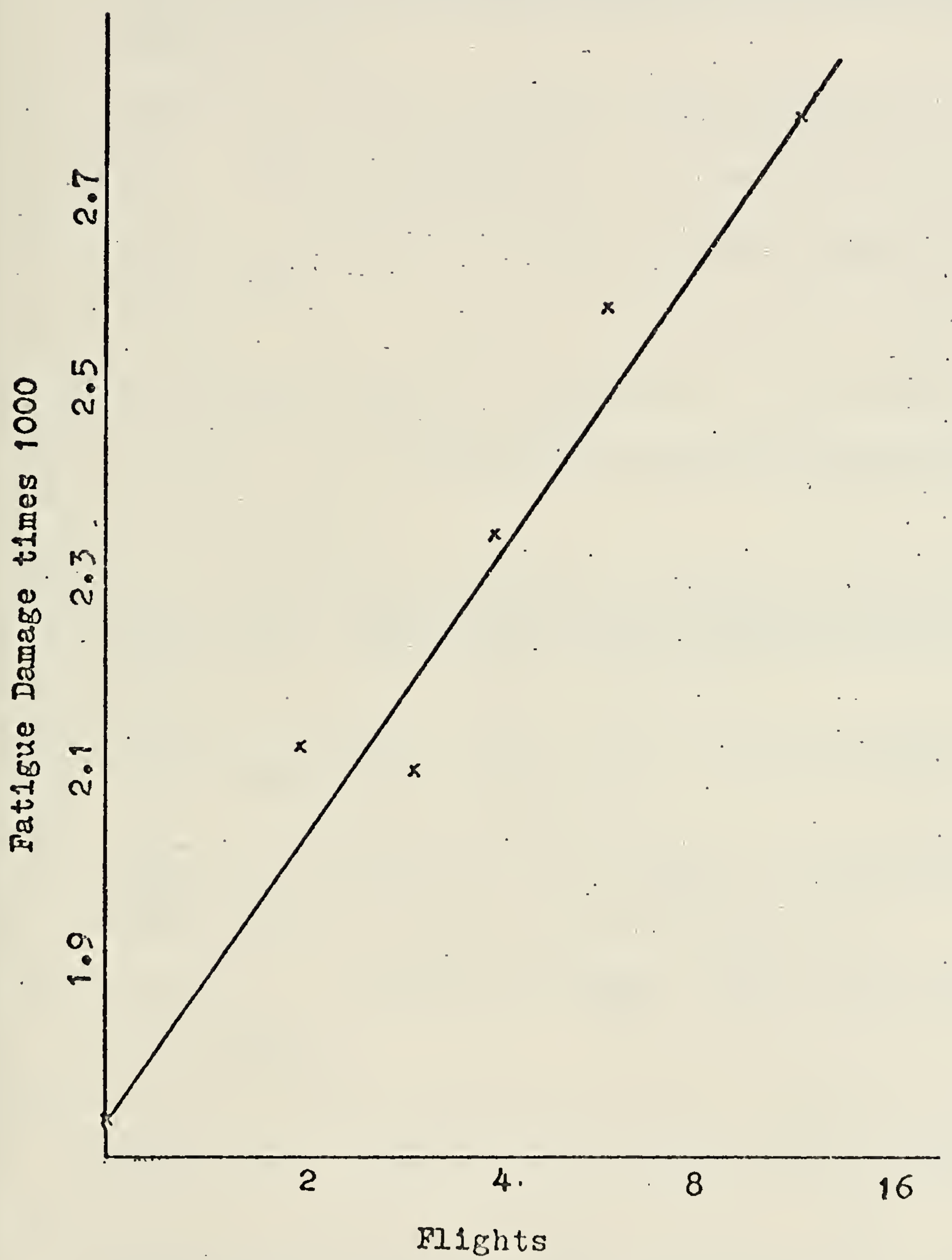

FIG.20 A-7B DAMAGE PREDICTION VS FLIGHTS (31.5 HRS.).

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